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CRITICAL REGIMES OF SUPERSONIC JET AIRCRAFT - STALL AND SPIN

by M. G. Kotik

"Mashinostroyeniye" Press
Moscow, 1967

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By M. G. Kotik

Translation of: "Kriticheskiye rezhimy sverkhzvukovogo samoleta - svalivaniye i shtopor"
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ABSTRACT: The results of theoretical and experimental studies of phenomena connected with stall and spin, and with the escape of aircraft from those regimes which involve complex nonstationary motion of an aircraft in space, are presented in this book. Particular attention is paid to the results of flight experiments (tests) of modern aircraft for stall and spin.

The conditions for supersonic jet aircraft entering critical (turbulent) regimes are examined, as well as the characteristics of stability and maneuverability at large angles of attack and the effect on these characteristics of a number of factors: angular velocities for banking aircraft, Mach number, flight altitude, etc. A classification and analysis of the stall and spin regimes are given, as well as the reasons for their occurrence and the specific features of the course of these regimes in modern aircraft. The causes and character of stall and entry into spin at high supersonic speeds and high flight altitudes (including the dynamic ceiling) are shown. Methods are developed for increasing the safety of flight at large angles of attack.

This book was intended for scientific workers, engineers, and pilots, as well as teachers and students of aviation institutes. 9 tables, 140 illustrations; bibliography: 43 entries.

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FOREWORD

In recent years, in connection with a substantial increase in the speeds and flight altitudes of supersonic aircraft, there has been observed a great variety of critical flight regimes involving stall and spin. These are dangerous flight regimes when, under normal flight conditions, premeditated entry into these regimes is prohibited. However, in some special cases, involuntary entry of an aircraft into such a regime is possible. Therefore, considerable attention is being devoted to the study of the specific features of flight of modern supersonic aircraft in critical regimes, and in regimes close to critical, and to the piloting methods required for escape from these regimes.

A knowledge of the particular features of flight and piloting in critical regimes, and an understanding of the physical essence of the causes which bring them about, are necessary to engineers and pilots connected with production, testing, and final adjustment, or with flight testing of supersonic aircraft. A deeper understanding of specific features of the behavior of such aircraft in critical regimes, the causes for their sideslip and for going into a spin, can aid pilots, not only in avoiding such regimes, but also in recovery, i.e., it can bring about an increase in flight safety. Therefore, in presenting the materials in this book, we will give special attention to a description of the physical picture of the phenomena.

Until the present, there have been no articles in which specific features of the flight and piloting of supersonic aircraft in critical regimes were examined in great detail. Recent publications on this theme have involved only certain individual problems. This book is an attempt to collect and present in rather complete form, the principal theories and experimental data related to the specific features of the behavior, maneuverability, stability, and technical procedures for piloting supersonic aircraft in flight regimes which are critical and near-critical.

This book gives the results of studies by Soviet and foreign scientists, engineers, and test pilots. Some results of theoretical, bench, and flight tests conducted by this author or with his participation are also given in the book. These tests deal with the effect of altitude and Mach number on the characteristics of maneuverability and stability of supersonic aircraft, and the development of piloting methods for reliable recovery of the aircraft from stall and spin, etc.

The methods for calculating the parameters, as well as the description of the causes and physical nature of new phenomena connected with the entry of supersonic aircraft into critical regimes which are presented in this book, are the first to be published in the literature.

The author would like to express his deep gratitude to M. L. Gallay, a Candidate in Technological Sciences and a Hero of the Soviet Union, who made a number of valuable comments in studying the manuscript. The author would be grateful for any comments and advice which might aid in improving this book. Please send them to "Mashinostroyeniye", Moscow K-51, Petrovka 24, U.S.S.R.

CONVENTIONAL SYMBOLS

V_{min} - Minimum flight speed (instrument speed for rectilinear flight without slipping into a $c_{y_{st}}$ regime); /5

$V_{min\ per}$ - Minimum permissible flight speed (instrument speed for rectilinear flight without slipping into a $c_{y_{per}}$ regime);

V_e - Safety flight speed (minimum instrument flight speed at which, for a given aircraft under the conditions of normal flight, it is still possible to carry out elementary maneuvers safely);

$\vec{\Omega} = \vec{\omega}_0 + \vec{\omega}$ - Vector of the resultant angular velocity for rotation of the aircraft;

$\vec{\omega}_0, \omega$ - Angular velocity for rotation of the aircraft in the original flight regime, and its increase in disturbed motion;

$\Omega_x, \Omega_y, \Omega_z$ - Projections of the vector $\vec{\Omega}$ on an axis in a system of coordinates connected with the aircraft: ox_1, oy_1, and oz_1 (angular velocities of roll, yaw and pitch, respectively);

$\vec{n} = \frac{\vec{R}_a}{G}$ - Load factor vector;

n_x, n_y, n_z - Projections of the vector $\vec{n}$ on an axis of a system of coordinates connected with an aircraft: ox_1, oy_1 and oz_1 (longitudinal, vertical, and lateral stresses, respectively);

α_{st} - Angle of attack for stall (angle of attack for the beginning of an aircraft's stall in normal flight);

α^*_{st} - Angle of attack for stall in an inverted flight (angle of attack for the beginning of an aircraft's stall during flight "upside down");

α_{cr} - Critical angle of attack (angle of attack for an aircraft in the regime $c_{y_{max}}$);

α^*_{cr} - Critical angle of attack in inverted flight (angle of attack for an aircraft in the regime $c_{y_{min}}$);

δ - Angle of deviation for the controls;

P - Force on the control knob (or control stick);

P_p - Force on the pedals (the difference in stresses applied by the pilot to the right and left pedals);

x - Movement of the control knob (stick);

x_p - Movement of the pedals;

J_x, J_y, J_z - Axial moments of inertia for an aircraft relative to the corresponding axes of the related system of coordinates;

I_x, I_y, I_z - Plane moments of inertia for the aircraft (relative to the planes y_1oz_1, z_1ox_1, and x_1oy_1 connected with the aircraft's system of coordinates, respectively); /6

$c_{y\,max}$ - maximum lift coefficient (maximum value for the lift coefficient when the altitude, Mach number, angle of slip and external configuration of the aircraft in normal flight are known);

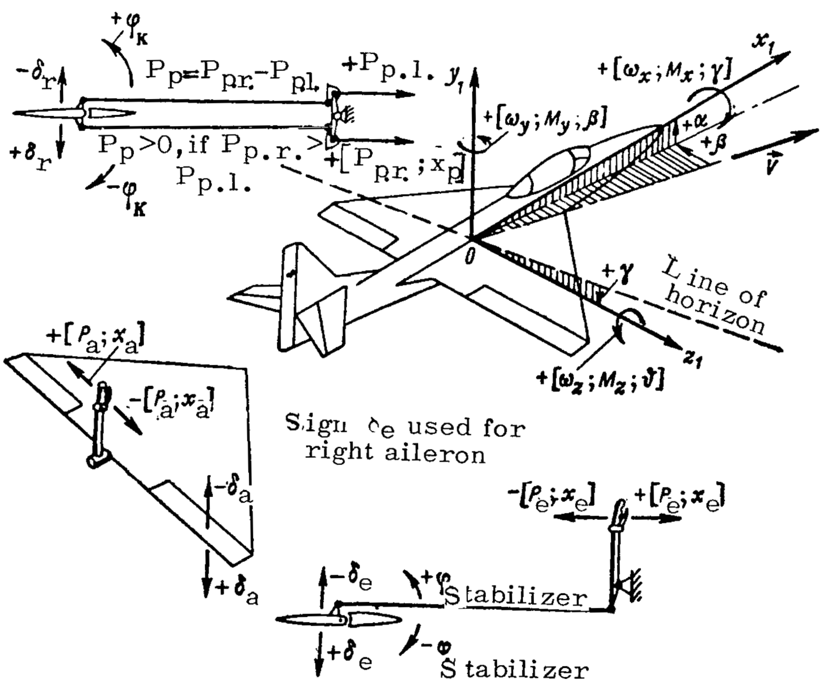

Fig. 0.1. System Designated for the Axes of the Coordinates Connected with the Aircraft, and Rule of Signs.

$c_{y\,min}$ - Minimum lift coefficient (maximum absolute value for the lift coefficient when the altitude, Mach number, angle of slip and external configuration of the aircraft in normal flight are known);

$c_{y\,st}$ - Lift coefficient during a stall (corresponds to the angle of attack for stall);

$c_{y_{jo}}$ - Lift coefficient for the beginning of a buffet (corresponds to the onset of buffet);

c_{y_δ} - Maximum admissible value for the lift coefficient at high supersonic flight speed (obtained by full deflection of the control stick backward);

$c_{y_{per}}$ - Maximum permissible value for the lift coefficient, according to the conditions for flight safety (restriction according to the buffet, according to the appearance of instability, etc) in normal flight of an aircraft. /7

The designations in this book for the correct system of related axes of the coordinates (All-Union State Standard 1075-41), and the rule of signs are shown in Figure 0.1.

In accordance with this, the signs of the vector components for load factor are determined as follows:

$n_x>0$ - if the pilot is pressed against the back of the seat (flight speed increases);

$n_x<0$ - if the pilot moves away from the back of the seat (flight speed decreases);

$n_y>0$ - if the pilot is pressed down into the seat;

$n_y<0$ - if the pilot is raised out of the seat;

$n_z>0$ - if the pilot leans to the left ($\beta<0$);

$n_z<0$ - if the pilot leans to the right ($\beta>0$).

INTRODUCTION

The flight regimes occurring at angles of attack equal to, or exceeding in absolute value, the angles of attack for the beginning of a stall are called critical, or, more precisely, critical in angle of attack (in the future, for the sake of brevity, we will call them simply critical or disturbed regimes). Such regimes are stall and spin. In a normal (not inverted) flight, these regimes correspond to angles of attack of $\alpha \geq \alpha_{st}$ while, in inverted flight (flight "upside down), they correspond to angles of attact of $\alpha \leq \alpha^*_{st}$ (Fig. 0.2). For certain aircraft, the angles of attack for stall coincide with the critical angles of attack:

$$\alpha_{st} = \alpha_{cr} \text{ and } \alpha^*_{st} = \alpha^*_{cr}.$$

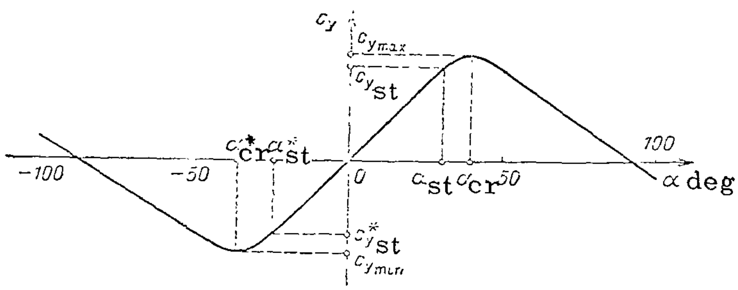

Fig. 0.2. Coefficient of Lift Versus Angle of Attack of the Aircraft (An Asterisk Indicates the Parameters of Inverted Flight, or Flight "Upside Down").

Stall and spin are not forms of piloting which are intentionally carried out under conditions of normal flight of an aircraft. They are non-cruising flight regimes for civil, as well as for military,aircraft. However, experiments with mass flight tests, and in particular, an analysis of aircraft emergencies and accidents, show that modern supersonic aircraft are no less likely to enter critical regimes than are subsonic aircraft. According to the foreign statistical data we have[1], out of all the aviation

[1]See footnote on page *xi*.

emergencies or accidents in recent years, about 15-20% were caused by entry of aircraft into these regimes.

The unintentional entry of supersonic aircraft into critical regimes can occur as follows: as a result of errors in piloting during flight at large angles of attack, at high original angular velocities of rotation of the aircraft or high supersonic Mach numbers of flight, and also, naturally, during an emergency (for example, the breakdown of the control system), or by the effect of incidental external factors (the action of a strong explosion wave, entry into the turbulent wake of an aircraft flying ahead, etc.). The specific features of the behavior, characteristics of stability and maneuverability, conditions and methods for piloting the aircraft in critical regimes, all differ basically from those in all other (cruising) flight regimes. The pilot's work during entry of an aircraft into critical regimes is made much more complicated.

The transition to high supersonic velocities and high flight altitudes has caused the appearance of substantial differences in the dynamics and piloting of supersonic aircraft, in comparison to subsonic speeds, in all the flight regimes, including critical regimes.

In order to increase flight safety, various aerodynamic design methods, gauges and signaling devices, flight restrictions, and piloting procedures are all being used, whose goal is to prevent the possibility of involuntary entry of an aircraft into critical regimes. If the aircraft nevertheless enters them, the pilot must have reliable methods for escaping such regimes. Unfortunately, there has as yet been no development of methods which guarantee the impossibility of involuntary entry of aircraft into these regimes under any conditions (including the most complicated flights - for example, during aerial combat, when the pilot is interested particularly in pursuing the enemy, and the conditions for combat maneuvering require that the aircraft emerge at great angles of attack.).

Therefore, recent examinations of problems related to the study of the specific features of flight and piloting of aircraft in critical regimes are of special interest. Of particular note is the problem of the necessity for a careful study of critical regimes and a development of methods for guaranteeing flight safety at large angles of attack, in view of the large number of commercial passenger aircraft flying at high altitudes. These problems have very important theoretical and practical significance.

During the first years of the birth and development of aviation, stall and the aircraft's going into a spin, as a rule, ended in an

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[1] W. Pleines: Probleme der Kurzstart Flugzeuge, (Problems of STOL Aircraft), Luftfahrttechnik, Vol. 8, No. 3, 1962.

accident (the lack of parachutes excluded the pilot's vacating the aircraft). We know of only isolated cases when the aircraft spun to the ground, and the pilot survived. This could occur, in particular, if the aircraft went into a spin with relatively large average values for the angle of attack, in a regime when the resultant velocity of the aircraft was relatively low. However, the existence of such particularly specific regimes was never suspected in those days. A large number of emergencies and accidents at that time were explained by the aircraft's entry into vortices in the air, although in fact (as became evident later on) the real cause was stall, with subsequent entry into spin.

Subsequently, according to the reports of pilots who had been saved after the aircraft went into a spin and the descriptions of eyewitnesses, the fact of occurrence and the character of such a complex and poorly-controlled flight regime were established. After this, theoretical and experimental studies of spin were begun. They were directed toward an explanation of the essence of this phenomenon, in order to work out recommendations for preventing the possibility of spin and stall, and also for finding methods for recovery of the aircraft from this regime when entry into it could not be prevented.

The first studies in this direction which provided mainly an understanding of the physical picture of the phenomena occuring during an aircraft's entry into spin, and recommended methods for recovery of the aircraft from a spin, were made in the Soviet Union by V. S. Pyshnov[2], and abroad by H. Glauert. Subsequently, Professor Pyshnov, an esteemed scientist and technologist and Doctor of Technical Sciences, conducted a number of detailed studies of the specific features of spin, the piloting procedures for pulling aircraft out of spin, and development of methods for calculating the spin characteristics of aircraft, which yielded important practical results. The studies on spin begun by Pyshnov and Glauert were then continued in studies by a number of scientists, test pilots, and engineers. The study of these dangerous flight regimes is still going on.

Studies of the critical flight regimes are being conducted by theoretical as well as experimental methods: analytical calculations, experiments with the aid of dynamically-similar models in wind tunnels, modeling of the regimes of stall and spin with real time simulators, calculations on adding machines, studies on free-flight models, and flight tests of real aircraft. Diverse studies on critical regimes, expanding our knowledge of the physical essence of the spin of subsonic aircraft, and developing methods for recovery from these regimes, were carried out by the esteemed sci-

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2 V.S. Pyshnov: Shtopor Samoletov (Spin in Aircraft). Trudy Voyennoy Vozdushnoy Akademii RKKA im. Zhukovskiy, Coll. 1, 1929.

entist and technologist, Professor A. N. Zhuravchenko[3], Doctor of Technical Sciences. Natural flight testing of an aircraft in spin was first done in the Soviet Union under the direction of Professor V. S. Vedrov, Doctor of Technical Sciences, with the talented aid of the test pilot and engineer, Yu. K. Stankevich[4].

A great contribution to the study of critical regimes was made by the following Soviet scientists: Professor V. F. Bolotnikov, Doctor of Technical Sciences, Professor B. T. Goroshchenko, Doctor of Technical Sciences, Professor G. S. Kalachev, Doctor of Technical Sciences, and Ye. A. Pokrovskiy, Candidate of Technical Sciences. Many important articles on the study of spin were written by Ya. I. Teteryukov, Candidate of Technical Sciences, Engineers M. M. Mikhaylov, V. P. Tulyakov and V. M. Zamyatin, and others.

Some very complex and rewarding flight studies and tests of aircraft in critical regimes could not have been carried out without the talented aid of highly-qualified test pilots, among whom we should single out: V. P. Chkalov, M. M. Gromov, Yu. K. Stankevich, A. N. Grinchik, Ya. I. Vernikov, S. N. Anokhin, G. A. Sedov, A. I. Nikashin, N. S. Rybko, V. G. Ivanov, L. M. Kuvshinov, A. G. Kochetkov, G. T. Beregoviy, A. A. Shcherbakov, V. F. Kovalev, V. S. Kotlov, O. V. Gudkov, and others.

[3] A. N. Zhuravchenko: Metody resheniya zadach shtopora i ustoychivosti, upravlyayemosti samoleta pri potere skorosti (Methods for Solving the Problems of Spin, Stability, and Maneuverability of an Aircraft during Loss of Speed). Trudy Tsentral. Aero-Dynam. Inst., No. 167, 1934.

[4] V. S. Vedrov, S. A. Korovitskiy, and Yu. K. Stankevich: Issledovaniye shtopora samoleta R-5 v polete (Studies on the Spin of an R-5 Aircraft in Flight). Trudy Tsentral. Aero-Dinam. Inst., No. 228, 1935.

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CHAPTER 1

CONDITIONS FOR THE ENTRY OF SUPERSONIC AIRCRAFT INTO CRITICAL REGIMES

1.1. Specific Features of Stability and Maneuverability, and the Effect of the Mach Number

(a) PRINCIPAL FEATURES OF STABILITY AND MANEUVERABILITY

The characteristics of stability and maneuverability of supersonic aircraft differ substantially from similar characteristics for subsonic aircraft. Even during flight at subsonic speeds in a supersonic aircraft, a pilot who has previously flown in subsonic aircraft at the same speeds experiences a significant difference. In addition, the significant expansion of the range of flight speeds, Mach numbers, flight altitudes, and angles of attack for supersonic aircraft has led to a great change in their characteristics, depending on the flight regime. Thus, for example, the characteristics of stability and maneuverability for an aircraft change greatly during transition from subsonic to supersonic flight speed, which is related to specific features of supersonic flow. /13*

A change in the stability and maneuverability naturally leads to a corresponding change in techniques for piloting the aircraft.

Let us examine the principal features of stability, maneuverability, and techniques for piloting supersonic aircraft in subcritical regimes, whose character is significant from the point of view of the possibility of the aircraft entering critical regimes. The most important of these features are:

(1) A decrease in the degree of static stability of an aircraft in flight at high supersonic Mach numbers.

(2) A possibility of loss in stability due to the effect of the interaction of the aircraft's longitudinal and lateral motions during rotation.

* Numbers in margin indicate pagination in original foreign text.

(3) A decrease in the effectiveness of the control system with an increase in supersonic Mach numbers of flight, and their very high effectiveness at subsonic velocities, which is necessarily connected with it;

(4) A sharp change in the necessary movement of the longitudinal control lever per unit load factor during transition through the range of sonic Mach numbers of flight. Tnis change is connected with an increase in the longitudinal static stability and a decrease in the effectiveness of the longitudinal control. /14

(5) Speed instability within the range of sonic Mach numbers of flight.

(6) A deterioration of the dynamic properties of the aircraft (characteristic of damping, etc) when $M>>1$.

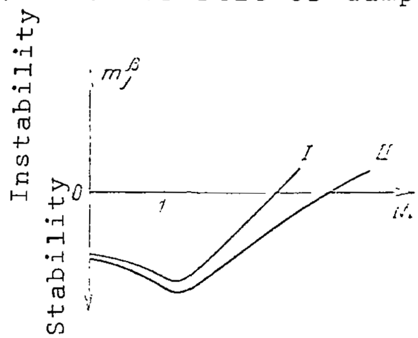


Fig. 1.1. Typical Function $m_y^\beta = f(M)$: I-Maneuver with load factor ($n_y > 1$); II-Rectilinear Horizontal Flight ($n_y = 1$).

The degree of the aircraft's static stability in flight, characterized by the derivative m_y^β, decreases greatly during transition to high supersonic flight speeds (Fig. 1.1.). This is caused mainly by a decrease in the effectiveness (supporting properties) of the vertical tail which is affected by the compressibility of the air, and also by the elastic deformation of the aircraft structure. The decrease in the degree of the aircraft's static flight stability produces the prerequisites for involuntary entry of the aircraft into critical regimes, for two reasons: first of all, because of the possibility of the appearance of relatively large sideslip angles, greatly decreasing the values for the coefficient of lift of the aircraft corresponding to the beginning of a stall; secondly, because of the possibility of the appearance (for a low degree of static stability in flight) of a sharp interaction between the longitudinal and lateral motions of the aircraft.

(b) INTERACTION BETWEEN THE LONGITUDINAL AND LATERAL MOTIONS OF AN AIRCRAFT DURING ITS ROTATION.

In speaking about the interaction of longitudinal and lateral motions of an aircraft, we usually have in mind the appearance of those interrelationships between the longitudinal and the lateral moments when the characteristics of lateral motion change spontaneously during change in some parameter of the longitudinal motion, and vice versa. In other words, during a change (for example) in the angle of attack together with a related change in the pitching moment (longitudinal moment), the yawing and rolling moments (lateral moments) also change simultaneously, and during a change in any characteristic of the lateral motion, there is a corresponding change (caused by this) in the parameters of the longitudinal motion.

Similar mutually related changes in the characteristics of longitudinal and lateral motions of an aircraft are determined by the aerodynamic, inertial, and gyroscopic moments and forces acting on it. Hence the cause for interaction of the longitudinal and lateral motions of an aircraft is the presence of aerodynamic, inertial, and gyroscopic interrelationships. In relation to this, there are three different types of interaction: aerodynamic, inertial, and gyroscopic. We will discuss briefly the physical significance of each of them. /15

AERODYNAMIC INTERACTION

Aerodynamic interaction includes, on the one hand, a change in the aerodynamic forces and moments causing the longitudinal motion of the aircraft, depending on the angle of sideslip, and (on the other hand) a change in the lateral aerodynamic forces and moments, depending on the angle of attack. Thus, for example, we know that the appearance of sideslip (to the right as well as to the left) produces a certain additional aerodynamic (subscript "a") moment of pitch ΔM_{za}. In modern supersonic aircraft, for small absolute values of the sideslip angle, this additional moment ΔM_{za} usually causes diving, and for very large sideslip angles, it usually causes pitching.

The aerodynamic moments of roll and yaw and their derivatives depend to a large extent on the value of the aircraft's angle of attack (we will speak about this in more detail later). This dependence is particularly strong for aircraft with sweptback wings at large angles of attack, when there are regions where the flow is partially separated at the wing tips. An unbalanced distribution of the regions of flow separation on the right and left halves of the wing (which, as a rule, occurs under real conditions because of aerodynamic or geometric asymmetry of the aircraft) can result in the appearance of significant additional aerodynamic moments of roll and yaw.

An important reason for the change in the aerodynamic forces and moments of one of the aircraft's motions (longitudinal or lateral) during a change in the parameters of its other motion is the kinematic link between the values for the angles of attack and sideslip with rotation of the aircraft. Thus, in particular, rotation of an aircraft relative to its longitudinal axis ox_1 (lateral motion) causes a change in the angles of attack (longitudinal motion) and sideslip, on whose values the aerodynamic moments and forces of the longitudinal and the lateral motions of the aircraft depend directly. For the sake of clarity, let us examine a simplified scheme for the change in the angles of attack and sideslip during rotation of an aircraft relative to its longitudinal axis, without considering the effect of the aircraft's weight and the corresponding changes in load factor on the character of the aircraft's motion (flight trajectory). The latter noticeably complicates the physical picture of the phe- /16 nomenon. However, it does not seem necessary to include that effect in order to understand the essence of the directly kinematic relation between the parameters mentioned above.

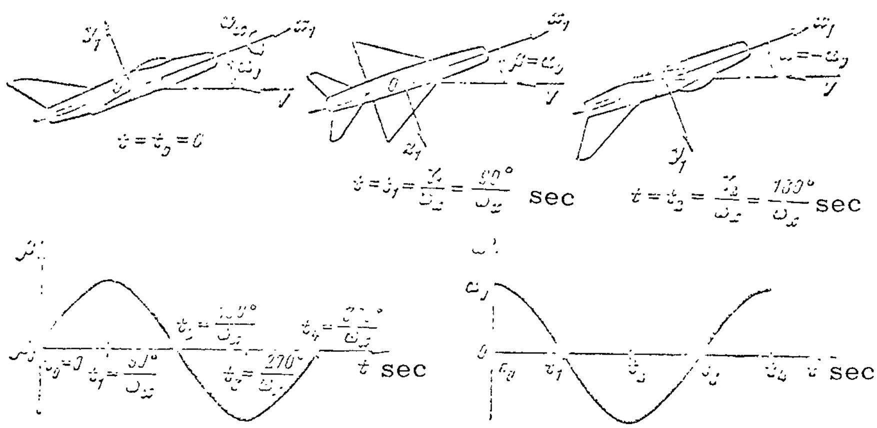


Fig. 1.2. Graph and Diagram Explaining the Kinematic Relation between the Angles of Attack and Sideslip, during Rotation of an Aircraft Relative to its Longitudinal Axis.

Let us assume that a model of an aircraft is placed in a wind tunnel and has only one degree of freedom--the possibility of rotating relative to the axis which coincides with the model's longitudinal axis ox_1. In this case, the angle between the velocity vector of the incident flow $\vec{V}$ and the model's longitudinal axis remains invariable during rotation. If, for example, the original (subscript "0") sideslip angle β_0 is equal to zero (Fig. 1.2; $t=t_0$). then during such rotation a turn of the model by 90° relative to its longitudinal axis results in a change in the angle of attack from its original value $\alpha_0(t=t_0)$ to zero $(t=t_1)$. Then during a turn through 180°, the angle of attack becomes equal to the original in absolute value, but has the opposite sign $(t=t_2)$. For a turn through 270°, the angle of attack again decreases to zero, then returns to its original value with a turn of the model through 360°.

In this case, the sideslip angle also changes in the same cyclic manner. For a turn of the model through 90°, the sideslip angle increases from its original zero value to a value equal to the original angle of attack $(t=t_1)$. After a turn through 180°, the sideslip angle again decreases to zero, and with a turn through 270°, it becomes negative and equal to the original angle of attack in absolute value (β* $-\alpha_0$). After a turn through 360°, the sideslip angle again becomes equal to zero, etc. Obviously, similar cyclic changes in the angles of attack and sideslip will occur during rotation of the model airplane relative to its vertical axis oy_1 (Fig. 1.3).

A visual representation of the interaction between the longitudinal and lateral motions of an aircraft in rotation can be obtained by examining the specific features of the motion of two hypothetical aircraft which have a certain original angular velocity of roll. We will consider conditionally that they rotate due to the effect of aerodynamic moments alone, and that they have no inertial moments. Let the first of them have infinitely large degrees of longitudinal and directional static stability, and let the second be statically neutral in the longitudinal and directional relationship, i.e., let the degrees of its longitudinal and directional stability be equal to zero. We will assume that these aircraft are in rectilinear horizontal flight with a certain original angle of attack α_0 without sideslip ($\beta_0 = 0$). /17

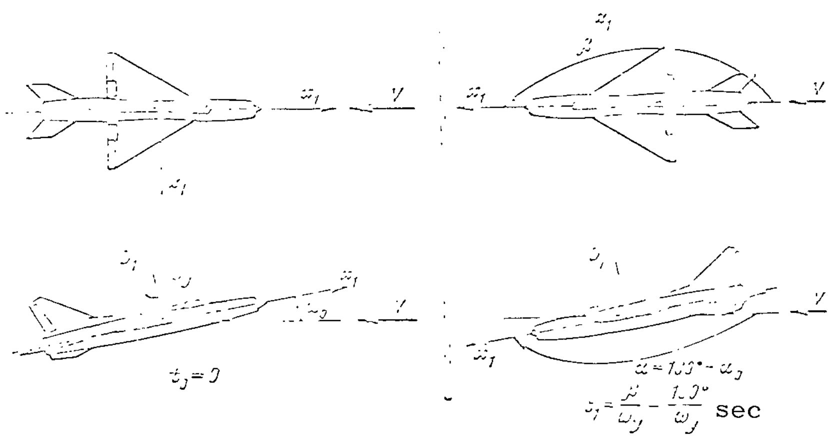


Fig. 1.3. Schematic Diagram Showing the Kinematic Relationship Between the Angles of Attack and Sideslip, during Rotation of an Aircraft Relative to its Vertical Axis.

If the moment of roll is added to the first of these two aircraft (tilting the ailerons), then the aircraft will begin to rotate without a change in the original values for the angles of attack and sideslip . Such behavior for the aircraft is caused by the infinitely large degrees of stability. In this case, during the change in the angles of attack and sideslip, there are reducing (stabilizing) aerodynamic moments which are infinitely large in absolute value, and which quickly return the aircraft to its original values for these angles.

For the sake of clarity in the discussions, we will simply consider that the center of gravity of such an aircraft will continue to move in a rectilinear trajectory during the appearance of the original angular velocity of roll ω_{x0} (if, for the sake of greater clarity, we disregard a bend in the trajectory because of the weight of the aircraft and the change in load factor during its rotation), and all the other points move in circles whose centers are located on the flight trajectory. These circles move along the trajectory /18 with the flight speed (Fig. 1.4). In this case, the aircraft's longitudinal axis forms a conic surface whose axis coincides with the direction (speed vector) of the flight. The same conic surface also moves with the flight speed.

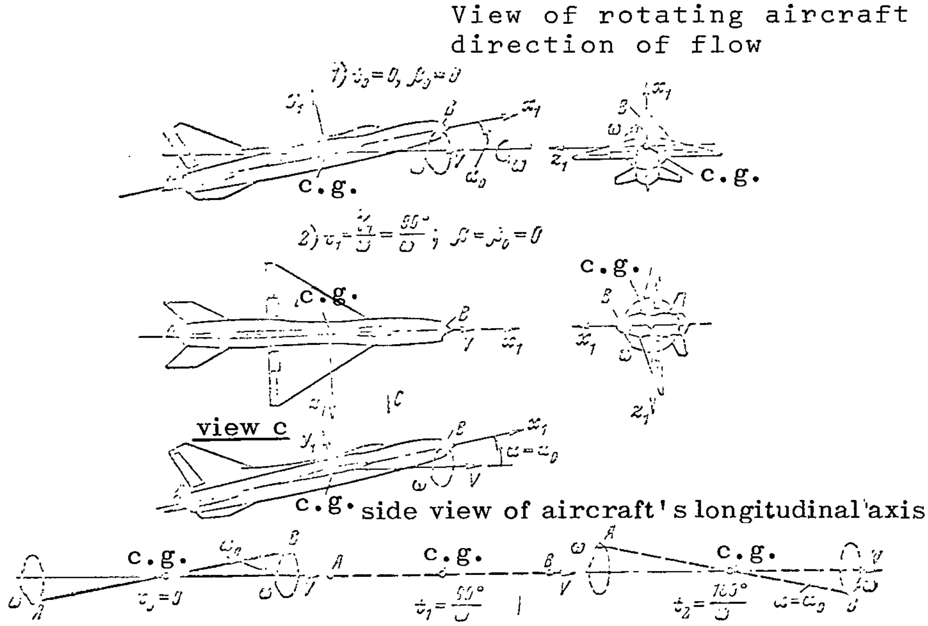


Fig. 1.4. Schematic Diagram Illustrating the Rotation of a Hypothetical Aircraft which has Infinitely Large Degrees of Longitudinal and Directional Static Stability.

If we add a moment of roll to the second (statically neutral) aircraft, it will then rotate relative to its longitudinal axis, moving in space parallel to itself (Fig. 1.5). This is explained by the fact that, due to the zero degrees of static stability, there will be no reducing aerodynamic moments during a change in the angles of attack and sideslip. Thus the pivot axis (longitudinal axis) of the aircraft will not move out of the vertical plane, and the angle between this axis and the flight trajectory will not change.

The degrees of longitudinal and directional static stability of a real aircraft have a definite finite value. Such an aircraft, when affected by a roll moment, will rotate relative to the axis O_ς, which does not coincide either with the direction of the flight trajectory (speed vector) or with the direction of its longitudinal axis (Fig. 1.6). This occurs because changes in the angles of attack and sideslip caused by rotation will be accompanied by the appearance of reducing aerodynamic moments which shift the aircraft's longitudinal axis from its original position. In this case, the angles of attack and sideslip will come closer to their original values as the degree of the aircraft's stability increases, and as the angular roll velocity decreases. If the latter has a small value, the aircraft will rotate relative to an axis which nearly /19

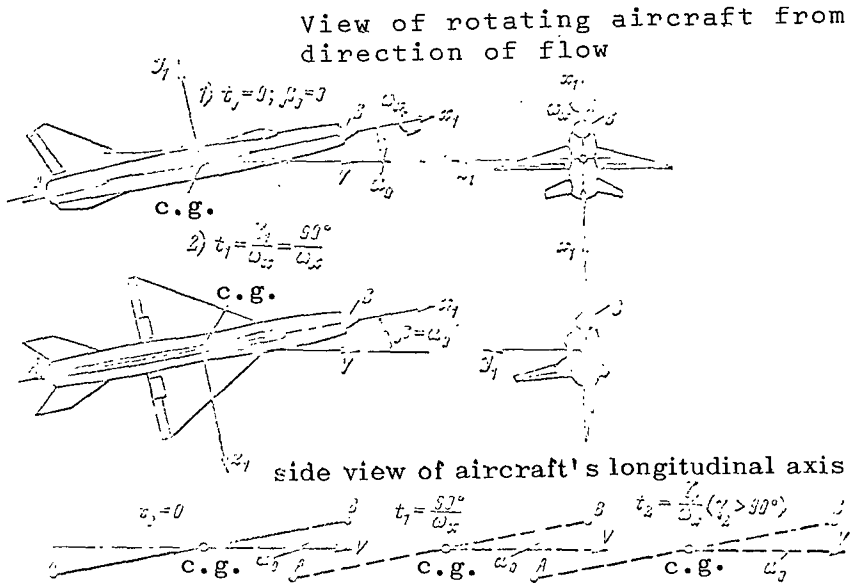


Fig. 1.5. Schematic Diagram Illustrating a Hypothetical, Statically Neutral Aircraft.

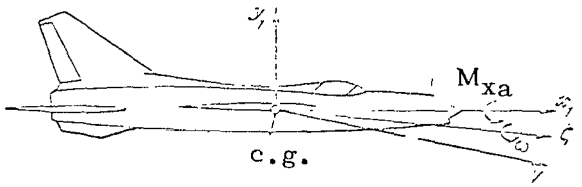


Fig. 1.6. Distribution of the Pivot Axis for a Real Aircraft, to which is added the Aerodynamic Roll Moment $M_x a$, Produced by Tilting of the Ailerons.

coincides with the direction of the flight trajectory. For a high angular velocity of the roll, the picture changes. The reducing /20 aerodynamic moments, in actual practice, will not succeed in eliminating the cyclic changes of the angles of attack and sideslip occurring during rotation, and the aircraft will rotate relative to an axis which nearly coincides with its longitudinal axis.

INERTIAL INTERACTION

If the inertial moments during rotation are negligibly small, then the aircraft will be stable in both of the cases examined above. However, real aircraft also have finite (very great for supersonic aircraft) inertial moments during rotation. The presence of inertial moments of pitch and yaw cause the appearance of the inertial interaction of the aircraft's motion, which naturally can occur only during rotation of the aircraft.

In the general case, an aircraft rotates about all three of its axes (longitudinal, vertical and lateral), i.e., if the angular velocities of roll, yaw, and pitch are non-zero. In other words, the aircraft in this case rotates about an axis which does not coincide with one of its three axes, and which is not located in the plane of the aircraft's symmetry. Such rotation in space will take place even when the pilot moves only the ailerons, which produce the original angular velocity of roll ω_{x0}, and holds the other control surfaces in their original balanced positions. In this case, by the effect of the spiral rotation and the rotation of the aerodynamic moments because of the deviation of the ailerons and of the gyroscopic moment produced by the rotation of the engine rotor, an angular velocity of yaw ω_y also appears.

The tilting of the ailerons and the gyroscopic moment also lead to the appearance of the angular velocity of pitch ω_z. The vector of the resultant angular velocity for rotation of the aircraft $\vec{\Omega}$ is equal to the sum of these three components, and can differ greatly from the value for the vector $\vec{\omega}_{x0}$. Rotation of the aircraft can occur, not only because of the formation of the original angular roll velocity ω_{x_0}, but also because of the original angular velocities of yaw ω_{y_0} and pitch ω_{z_0}: by moving the control surfaces, raising or lowering the flaps, etc. In these cases, the inertial interaction can be very strong.

The appearance of an inertial moment of pitch (longitudinal motion) when there is angular velocity of roll (lateral motion) causes an inertial interaction between the aircraft's longitudinal and lateral motions, and the appearance of an inertial moment of yaw during rolling of the aircraft produces an inertial inter action between its directional yaw and roll motions. The inertial interaction during rotation of an aircraft relative to its longitu- /21
dinal axis is a very clear example. Therefore, in analyzing inertial interaction, it is usually examined during the presence of only angular velocity of roll, whose maximum values, as a rule, greatly exceed the maximum values for the angular velocities of yaw and pitch. However, such a simplification sometines can lead to a significant distortion of the general picture of the motion of the aircraft. In such cases, we should examine the effect of all three components of the resultant angular velocity for the aircraft's rotation. In particular; the inertial interaction of the aircraft's motions in critical regimes is characterized by the usually noticeable effect of all three components of angular velocity.

The physical essence of the inertial interaction can be explained in the following way. Let us represent the distribution of all the masses of an aircraft simply, in the form of four concentrated masses $m_i (i = 1, 2, 3, 4)$ located in the plane $x_1 o z_1$.

Figure 1.7 is a schematic diagram illustrating the interaction between the inertial disturbing moment $M_{y_{in}}$ and the aerodynamic reducing moment M_{y_a} of yaw during rotation of the aircraft with angular velocity relative to an axis which coincides with the speed vector V. In this case, the moments of inertia for the aircraft

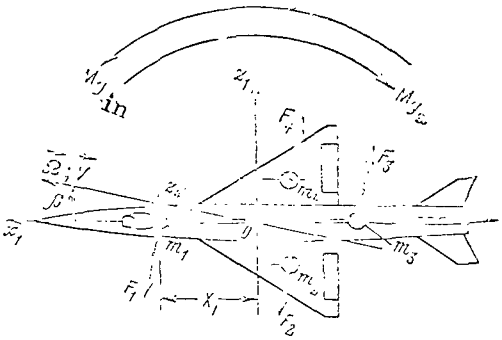


Fig. 1.7. Schematic Diagram Illustrating the Interaction of Moments of Yaw during Rotation of an Air-Aircraft About an Axis which Coincides with the Speed Vector: $m_1, \ldots, m_4$ are the Masses of Parts of the Aircraft; $F_1, \ldots, F_4$ are the Corresponding Centrifugal Forces.

relative to its axes ox_1, oy_1, and oz_1 are equal to:

$$\left.\begin{aligned} J_x &= \sum_{i=1}^{4} m_i z_i^2, \\ J_y &= \sum_{i=1}^{4} m_i (x_i^2 + z_i^2), \\ J_z &= \sum_{i=1}^{4} m_i x_i^2. \end{aligned}\right\} \qquad (1.1)$$

From this, we can see in particular that for the given assump- /22
tions,

$$J_y = J_x + J_z \qquad (1.2)$$

For the sake of simplicity in the discussions, we will consider that the aircraft rotates relative to the vector of the flight speed $\vec{V}$, i.e., that the directions for the vectors $\vec{V}$ and $\vec{\Omega}$ coincide. In this case, the centrifugal force of inertia F_1 during an aircraft's rotation at angular velocity Ω is determined by the expression:

$$F_1 = m_1 \Omega^2 x_1 \sin \beta,$$

where β is the original sideslip angle.

The inertial moment of yaw M_{y1} caused by this force is equal to

$$M_{y_1} = F_1 x_1 \cos \beta = m_1 \Omega^2 x_1^2 \sin \beta \cos \beta = \frac{1}{2} m_1 \Omega^2 x_1^2 \sin 2\beta.$$

Having obtained similar expressions for the moments produced by the remaining three masses, and using the equation in (1.1), we can find the resultant inertial moment of yaw in the following form:

$$M_{y_{in}} = \sum_{i=1}^{4} M_{y_i} = \sum_{i=1}^{4} \frac{1}{2} m_i \Omega^2 x_i^2 \sin 2\beta - \sum_{i=1}^{4} \frac{1}{2} m_i \Omega^2 z_i^2 \sin 2\beta = \frac{1}{2} \Omega^2 (J_z - J_x) \sin 2\beta. \tag{1.3}$$

A schematic diagram illustrating the interaction between the inertial disturbing moment $M_{z_{in}}$ and the aerodynamic reducing moment M_{za} of pitch during rotation of an aircraft with angular velocity Ω relative to the axis which coincides with the speed vector V is shown in Figure 1.8. Having examined Figure 1.8, we can easily find the corresponding expression for the inertial moment of pitch:

$$M_{z_{in}} = \sum_{i=1}^{2} m_i \Omega^2 x_i^2 \cos\alpha \sin\alpha = \frac{1}{2} \Omega^2 J_z \sin 2\alpha = \frac{1}{2} \Omega^2 (J_y - J_x) \sin 2\alpha. \tag{1.4}$$

We can see from formulas (1.3) and (1.4) that the inertial moments for given values of α and β increase with an increase in the angular velocity and weight distribution along the aircraft's longitudinal axis.

An increase in the weight distribution along the aircraft's lateral axis results in a decrease in the inertial moments.

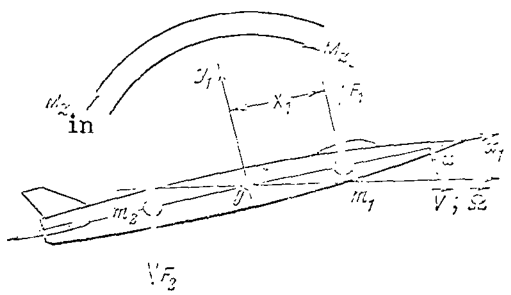


Fig. 1.8. Schematic Diagram Illustrating the Interaction between the Moments of Pitch during Rotation of an Aircraft Relative to the Axis which Coincides with the Speed Vector: m_1 and m_2 are the Masses of the Aircraft; F_1 and F_2 are the Centrifugal Forces. /2

Supersonic aircraft have a long fuselage and large weight distribution along the axis for a relatively small wingspan. Because of the significant flight speeds and the relatively weak roll damping, these aircraft can have high angular velocities of roll. Therefore, during their rotation it is possible to have great inertial moments of yaw and pitch. These moments disturb and reduce the stability, and in certain cases they can even lead to instability of the aircraft.

GYROSCOPIC INTERACTION

The gyroscopic interaction between the aircraft's motions, like the inertial interaction, shows up only during its rotation. The gyroscopic moment of the engine rotor appears during rotation of the aircraft relative to an axis which does not coincide with the axis of rotation of the rotor. The value and direction of the vector for the gyroscopic moment M_{gyr} are determined by the expression:

$$\vec{M}_{gyr} = J_p [\vec{\omega}_p, \vec{\Omega}], \qquad (1.5)$$

where

J_p is the polar moment of inertia for the engine rotor;
ω_p is the angular velocity for the rotation of the engine rotor.

It follows from this, in particular, that if the vector $\vec{\omega}_p$ is directed along the longitudinal axis of the aircraft ox_1, then when there is an angular velocity of the yaw Ω_y (lateral motion), for example, there is a gyroscopic moment of pitch (longitudinal motion). Thus there is a gyroscopic interaction between the longitudinal and lateral motions of the aircraft. Depending on the direction of rotation of the engine rotor, the gyroscopic moment M_{gyr} can be stabilizing or disturbing.

BEHAVIOR AND SPECIFIC FEATURES OF PILOTING AN AIRCRAFT DURING INTERACTION OF MOTIONS /24

If the angular velocity of the aircraft's rotation increases to such a degree that the sum of the disturbing inertial and gyroscopic (for the direction corresponding to the rotation of the engine rotor) moments during rotation becomes greater than the sum of the aerodynamic restoring (caused by the aircraft's static stability) and damping moments, there is a loss in stability. This results in a spontaneous increase in the absolute values for the angles of attack and/or sideslip of the aircraft, and thus in a spontaneous increase in the load factor.

The interaction of an aircraft's motions during its rotation can lead to the appearance of the so-called inertial (or, more precisely, aero-inertial) auto-rotation of the aircraft. For a pilot, inertial autorotation of an aircraft (according to the sensations he feels) means aerodynamic self-rotation causing the appearance of a spin regime. However, these two types of auto-rotation are actually very different. Aerodynamic auto-rotation occurs only at supercritical angles of attack, as a result of a loss in the aerodynamic roll damping of the aircraft, while inertial auto-rotation occurs even at subcritical angles of attack. The rolling static stability of an aircraft has a very important role in the development of inertial auto-rotation.

The effect of the moment of roll static stability on the disturbed motion of an aircraft depends on the value of the original angle of attack α_0, which determines the character and degree of sideslip development while the ailerons are tilted. For a positive original angle of attack (see Fig. 1.2, $t=t_0$), the aircraft's rotation caused by tilting the ailerons results in a glide with one wing lowered (see Fig. 1.2, $t=t_1$). Because of the effect of the aircraft's rolling static stability, there is an additional moment of roll which prevents rotation. If the original angle of attack is negative, the aircraft's rotation by tilting the ailerons results in a sideslip on the raised wing: sideslip toward the side which is opposite the aircraft's rotation. This causes the appearance of a supplementary moment of roll, which tends to increase the angular velocity produced by tilting the ailerons.

The disturbing inertial moment of pitch occurring under the influence of rotation tends to increase the absolute value of the negative angle of attack, which then causes an increase in the glide angle during rotation and so further increases the additional moment of roll produced by the rolling stability, etc. As a result, for a certain combination of parameters for the aircraft's motion, there is inertial auto-rotation. In this case, the resultant angular velocity of roll can be so great that even a full tilt of the ailerons to the side opposite to the rotation does not terminate it. Inertial auto-rotation is usually accompanied by high-amplitude vibrations due to changing load factor ("drops").

An example of spontaneous increase in the absolute load factor value during an aircraft's rotation is given in Figure 1.9.

The graph shows records from recording instruments, obtained during flight in a supersonic jet aircraft, which characterize the changes with time of the vertical (n_y) and lateral (n_z) load factors, the angular velocity of roll (ω_x), and the angle of tilt for the ailerons (δ_a). We can see from the graph that, in the case being examined, there is a sharp increase in the absolute values of n_y and n_z with an angular velocity of roll. In particular, the change in the value of n_y in relation to the original flight regime was

$$\Delta n_y \approx -1.5 - 2 = -3.5$$

The greatest absolute value for the angular velocity of roll in this regime was $|\omega_x| = 5$ rad/sec. An example of inertial auto-rotation in an aircraft is shown in Figure 1.10 (records made by recording instruments in flight). As we see from the records, after tilting the ailerons there was a great angular velocity of roll whose absolute value gradually exceeded 5 rad/sec. The records show that this angular velocity of roll changed little even after almost complete deviation of the ailerons (at the 18th second) toward the side which was opposite to the rotation of the aircraft.

In the regime shown in Figure 1.10, the vibrational amplitudes for vertical (n_y) and lateral (n_z) load factors reach values of ≈ 2.5 and ≈ 2.0, respectively.

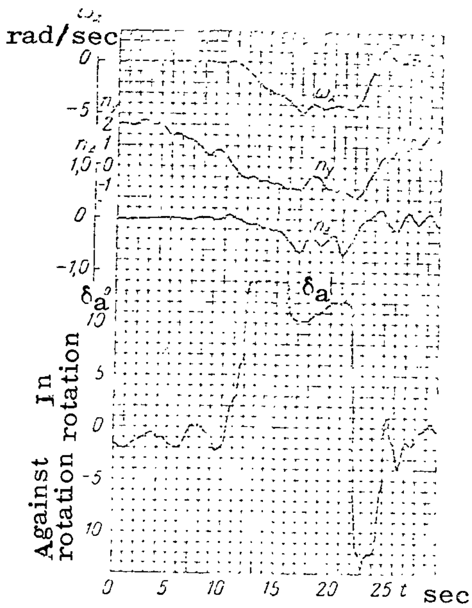
In practice, the most probable entry of an aircraft into regimes with a sharply-expressed interaction of motions during formation of a great angular velocity of roll (on the order of $|\omega_{x_0}| = 1.5\text{-}2.0$ rad/sec and more) is found in the process of the aircraft's carrying out maneuvers with an intensive change in the vertical load factor particularly when there is initial sideslip. In particular, as reported in the foreign press, such flight regimes occurred in supersonic American aircraft, the North American F-100 "Super Sabre", when its pilot made an energetic pull-out and simultaneously tilted the ailerons for a turn, particularly if the pilot in thes case did not keep the pedals in a neutral position. In some cases, the entry of F-100 aircraft into these regimes ended in an accident.

Piloting the aircraft during the appearance of interaction between the longitudinal and lateral motions, caused by its rotation, greatly complicates the situation, since the actions of the control surfaces to which a pilot is accustomed do not usually lead to the results he desires. Moreover, in some cases the aircraft's reaction in these regimes to deviation of the control surfaces can even prove to be the reverse, with an opposite reaction in all flight regimes. /26
Thus, for example, in these regimes a pilot's attempt to counter the increase in the absolute value of the negative vertical load factor by pulling the control stick back, as a rule, only intensifies the aircraft's rotation and causes even more intense increases in the load factor.

As pilots' experience has shown, in order to pull an aircraft out of a regime in which there is inertial interaction, the stabilizers are usually set in a neutral position. However, the forces acting on the pilot can prevent this to a great degree, as can great forces on the pedals in a guidance system without boosters (particularly at high supersonic Mach numbers or low altitudes during high instrument flight speeds).

For each aircraft in which it is possible to go into flight regimes with sharp interaction between the longitudinal and lateral /27
motions, a corresponding piloting method has been developed. A piloting method for pulling an aircraft out of regimes with such motion interaction is first developed in models, and then the method selected is checked in flight tests.

For a pilot, the principal signs of the approach of a regime with the appearance of inertial interaction of motions are the following: intense increase in load factor, which does not correspond to the pilot's sensations for the given position of the aircraft, the flight regime, and the position of the control surfaces; a sharp increase in the angular velocity of the roll, which does not correspond to the tilt of the ailerons in a given original flight



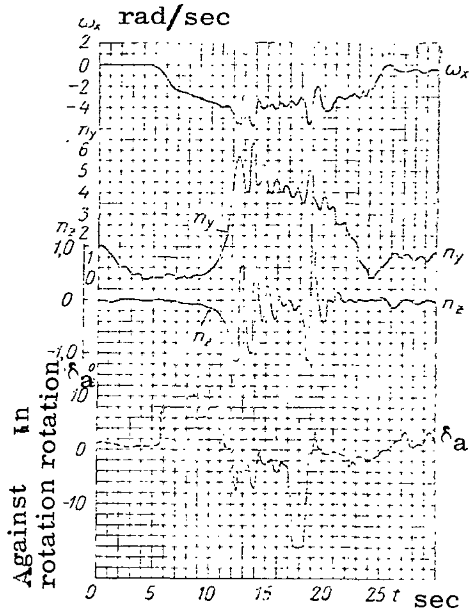


Fig. 1.9 Fig. 1.10

Fig. 1.9. Spontaneous Increase in Load Factor as a Result of the Interaction Between an Aircraft's Longitudinal and Lateral Motions (Records Made by Recording Instruments in Flight).

Fig. 1.10. Change in Load Factors n_y and n_z with Time During Inertial Autorotation of an Aircraft (Records Made by Recording Instruments in Flight).

regime; for aircraft with boosterless guidance systems, a spontaneous movement of the pedals and a sudden unusual increase in the forces on them, etc. If the pilot detects these signs in good time and turns the control surfaces and ailerons into a neutral position by an energetic action, then the aircraft will not go into such a regime. In the opposite case, i.e., when the pilot moves the rudders incorrectly, imprecisely, or not in time, a similar regime may occur. As a result, the aircraft can go into critical regimes (enter a stall), or $n^t_{y\,max}$ will be exceeded (the greatest permissible value, for structural strength, for load factors under the conditions of a normal test flight in an aircraft.

A theoretical analysis shows that, in a given aircraft, when the wing loading of a wing (G/S= const), the altitude (H = const). and the Mach number (M = const) for the flight are given, the danger of the undesirable appearance of an interaction between the longitudinal and lateral motions (the danger of losing stability) basically depends on a combination of three factors: the degree of longitudinal ($m_z^{c_y}$) and directional (m_y^{β}) static stability for an aircraft and the value for the angular velocity of its rotation. A visual representation of the critical (from this point of view) flight conditions gives the graph for the boundaries of stability shown in Figure 1.11.

We can see from the graph that the "corridor" formed by the envelope factors (I) of the sets of hyperbolas showing the boundaries for the stability ranges, obtained for various values of the angular velocities for the aircraft's rotation, expands with an increase in the angular velocity of the roll. This expansion of the corridor is caused by the presence of damping of yaw and pitch.

If an imaginary point $(|m_z^{c_y}|, |m_y^{\beta}|)$ is located within this corridor, then with these conditions for the flight regime, the aircraft will be stabilized in the rolling motion (when the aircraft rolls) for any value of the angular velocity. If the imaginary point is located outside this corridor, the aircraft will lose /28
stability for a certain value of the angular velocity of the roll. Thus, for example, let the flight conditions being examined correspond to an imaginary point A (See Fig. 1.11). This point is located within the range (close to its boundary) marked out by the stability boundary which corresponds to an angular velocity for an aircraft's rotation of 2 rad/sec. Subsequently, the aircraft loses stability in a given regime because of the motion of the roll when it reaches an angular velocity somewhat less than 2 rad/sec.

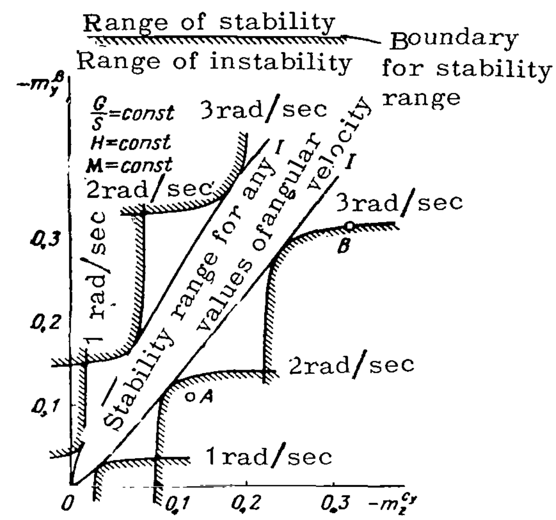


Fig. 1.11. Boundaries for Stability in the Motion of a Rotating Aircraft: I-the Envelope Factor for the Boundaries of the Stability Ranges

The aircraft whose imaginary point is Point B will maintain stability for angular velocities of roll less than 3 rad/sec. With the appearance of an angular velocity of roll equal to 3 rad/sec in absolute value, the aircraft in the given flight regime loses stability. With the appearance of angular velocities exceeding 3 rad/sec, the aircraft becomes unstable.

(c) DECREASE IN THE EFFECTIVENESS OF THE CONTROL SYSTEM.

A decrease in the effectiveness of an aircraft's control system during transition to supersonic Mach numbers of flight results in greater effectiveness at subsonic velocities. From the example given in Figure 1.12, we can see that the absolute value for the coefficient of the stabilizer's effectiveness m_z^{φ} (longitudinal control of the aircraft) during transition from subsonic to high supersonic Mach numbers decreases by a large factor. The decrease in $|m_z^{\varphi}|$ can also be caused by the appearance of aeroelasticity (in particular, elastic deformations of the fuselage) and by a

number of other effects. The effectiveness of the elevator, during transition to supersonic flight speeds, decreases much more sharply than the effectiveness of the stabilizer, which is explained by the impossibility of propagating disturbances upstream in supersonic 29
flow.

In a supersonic flow, movement of the elevator causes a change in pressure (an increase in the lift of the horizontal empennage) almost on itself alone, and this does not affect the pressure distribution of the fixed stabilizer located ahead of it. Therefore, a movable stabilizer is used instead of an elevator in modern supersonic aircraft. The movable stabilizer's effectiveness at subsonic velocities is very great (because of the necessity of preserving enough of its effectiveness at high supersonic velocities). For these aircraft, in flight at low altitudes with high subsonic instrumental velocities (high-speed thrusts), it would seem disadvantageous (with such a high effectiveness of the movable stabilizer) to have a combination of frequency characteristics for the aircraft and for the pilot. As a result of this, the "aircraft-pilot" system in some cases would even lose stability, and there would be an involuntary (for the pilot) longitudinal buildup of oscillations for the aircraft, accompanied by considerable oscillation of the vertical load factor n_y. Such motions of the aircraft can occur in the process of carrying out maneuvers which require precise piloting, if in this case the period of natural longitudinal oscillations of the aircraft is short (on the order of 1 sec; but this, as a rule, takes place in modern light aircraft in flight regimes with high-speed thrusts), and we can see a retardation of the pilot's movements and a slack in the control line (which acts quicker now in modern aircraft, because of boosters). The latter can cause increased sensitivity of the aircraft to considerable deviations of the longitudinal-control lever (system).

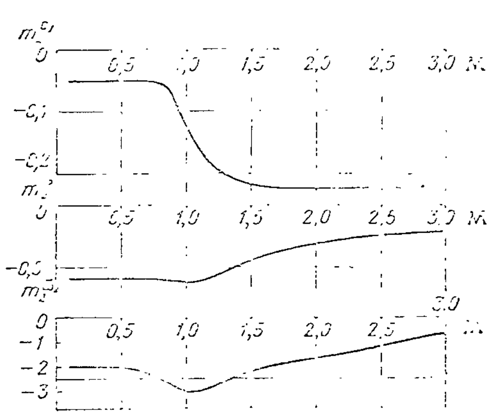


Fig. 1.12. Change in the Aerodynamic Derivations $m_z^{c_y}$, m_z^{ϕ}, and $m_z^{\bar{\omega}_z}$ for the Mach numbers of Flight in a Modern Supersonic Aircraft.

Thus, the principal causes for the "oscillation" are: the aircraft's increased sensitivity to small movements of the control lever (almost uncontrollable by the pilot), and short periods of natural oscillations of the aircraft, i.e., extremely light and superfluously sensitive control for small forces (necessary for piloting) on the level and small deviations /30
of it, as combined in a noticeable lag in the pilot's actions and a lag in the control system. An increase of friction and slack in the control system aggravates this situation and complicates "damping" with the necessary deviations of the rudder, because of a deterioration in control over the control stick and the appear-

ance of "instability zones" in the longitudinal control system.

In modern supersonic aircraft, in order to overcome the buildup of oscillations, the characteristics for maneuverability are selected so that there is no allowance for this occurrence in any of the test-flight regimes. With this goal in particular, special automatic devices are installed in the longitudinal control system: for example, devices for regulating the forces on the control sticks (ARS), devices for regulating the gear ratios for the control system and the force (ARC--the so-called control devices), dampers of the longitudinal vibrations of the aircraft, etc. However, in the case of breakdown or incorrect operation of such a control system, the aircraft can go into this disadvantageous, and sometimes even dangerous, regime. If for any reason the aircraft did go into a regime of vibration buildup, then it is not possible to counter each individual vibration by deviations of the control lever, since this only complicates piloting and can result in the appearance of even greater vibrations of the aircraft (it is difficult for the pilot to "deteriorate" all the vibrations of the aircraft). In such a case, it is necessary to reduce speed, holding the control lever in a certain middle position (it would be best to have it in a neutral, or close to neutral, position, moving it slightly in order to decrease the flight speed).

The effectiveness of the directional and roll control systems also decreases greatly during transition to high supersonic flight speeds. During flight at low altitudes, at high subsonic speeds, there can sometimes be lateral vibrations of the aircraft, accompanied, in particular, by considerable vibrations of the lateral load factor n_z (under conditions similar to those for longitudinal vibrations). In order to counter the lateral vibrations, the pilot must decrease the flight speed, keeping the pedals in a neutral position in this case.

(d) CHANGE IN THE MOTION OF THE CONTROL STICK PER UNIT LOAD FACTOR

The motion of the longitudinal control stick per unit load factor x_B^n can change greatly while passing through ranges of sonic flight speeds (Fig. 1.13). The change in the value for x_B^n is explained mainly by the decrease in the control system's effectiveness and by the increase in the degree of longitudinal static stability of the aircraft during transition to supersonic speeds, and in some cases by a number of other causes also (in particular, by elastic deformations of the fuselage). When carrying out maneuvers while /31 decelerating the aircraft to sonic speeds, this can lead to the aircraft's "pitching up" if the pilot does not succeed in moving the control stick in time from the position corresponding to the conditions for supersonic flight speeds.

The pitchup process usually occurs rather slowly, and the pilot can succeed in countering it by a smooth motion of the control stick away from himself. However, in a most disadvantageous coin-

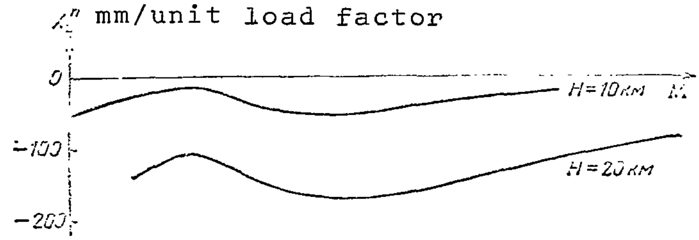


Fig. 1.13. Change with Mach Number of the Motion of the Control Stick per Unit Load Factor.

cidence of circumstances (for example, in the case of intense deceleration with a non-operational engine and disabled flaps, for a climb with minimum rear centering), the pitchup can be very intense and may be accompanied by a sharp refection of the force. If the flight occurs at high altitudes (large angles of attack for rectilinear horizontal flight), then this also causes an aircraft to enter critical regimes when the pilot's actions are not sufficiently precise.

(e) INSTABILITY WITH SPEED AND OTHER FEATURES

Speed instability in the range of sonic Mach numbers of flight (so-called "dip in speed" in the type of balancing curve shown in Fig. 1.14) is caused by the appearance of diving aerodynamic moments as a result of shifting the aircraft's aerodynamic center rearward, a decrease in the downwash in the region where the horizontal section of the empennage is located during an increase in c_y (with an increase in the rate of thrust), by the appearance of supersonic flow ranges, and also by a number of other effects. For an example, Figure 1.14 shows balancing curves for moving the aircraft's control stick $X_B = f(M)$ in a rectilinear horizontal flight without glide, for two altitudes. The speed instability range is characterized by the negative slope of the balancing curves (segment *a-a*, for which the curve slopes down).

Speed instability somewhat complicates piloting the aircraft, although this in itself, for normal operational conditions, does /32
not present any great difficulties to the pilot since it occurs, for example, in the case of instability in

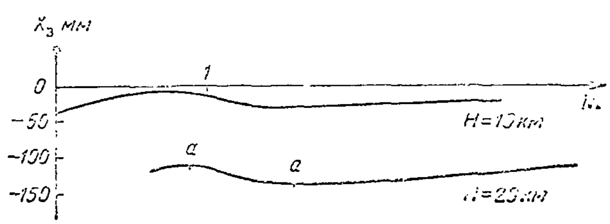


Fig. 1.14. Trim Stick Position Versus Mach Numbers.

In addition to those we have mentioned, there are many other features of stability and maneuverability in supersonic aircraft which cause them to enter critical regimes under certain conditions. We will mention some of these features which exist to a greater or lesser degree in all supersonic aircraft. Thus, for example, a decrease in the degree of an aircraft's directional static stability, a weakening of the damping properties for the tail empennage and the wing because of a decrease in the growth of the coefficients for lift and lateral aerodynamic forces on the degrees of change in the angles of attack and glide (a decrease in c_y^α and c_z^β) for high supersonic M numbers, leads to deterioration of the damping of the aircraft's disturbed motion, an increase in the oscillations of the force - an important characteristic for the transitional processes (transient flight regimes), i.e., the aircraft's dynamic maneuverability.

Fig. 1.15 shows an example of the change of a normal load factor with time during an aircraft's disturbed motion after a sharp movement of the elevator. We can see from the graph that oscillation of the load factor $\Delta n_{stress} = n_{stress} - n_{est}$ can reach substantial values. This is very clearly indicated in the characteristics of the aircraft's maneuverability and the pilot's sensations in carrying out the maneuvers[1]. In some cases, oscillation of the load factor can be the cause of the aircraft entering critical regimes. The established value for a normal load factor n_{est} shown in the graph differs from the original value n_{orig} by the value $\Delta n_{est} = n_{est} - n_{orig}$.

The decrease in the aerodynamic derivatives c_y^α and c_z^β, the increase in the wing loading (ration between the aircraft weight and the area of the wing), and the flight speeds lead to the circumstance that, for supersonic aircraft at high Mach numbers, it is necessary to increase the angles of attack and sideslip by a larger factor in order to obtain the same speed for a change in the angles of the slope for the trajectory (the direction of the flight), than for subsonic aircraft. This also causes entry into regimes with critical angles of attack (particularly at average and relatively low instnument flight speeds, i.e., at high altitudes). /33

The presence of external structures under a wing or on the fuselage of an aircraft (for example, suspended fuel tanks) usually results in a foreward displacement of the location of the resultant lateral aerodynamic force, which decreases the degree of directional static stability somewhat. The transfer of the aircraft's center of gravity toward the fromt and the increase in the thrust of the turbojet engine (P) also causes a decrease in the directional stability. The latter can be explained in the following way.

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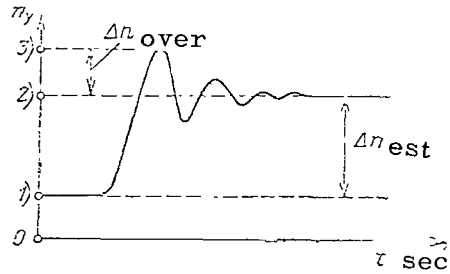
1 For more detail on this, see for example M.G. Kotik, A.P. Pavlov, et al.: Letnyye ispytaniya samoletov (Flight Testing of Aircraft). Mashinostroyeniye, 1965.



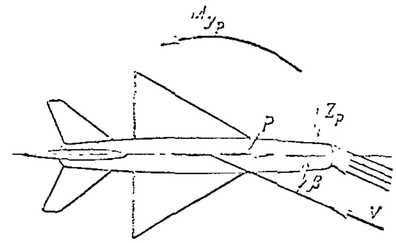


Fig. 1.15. Fig. 1.16.

Fig. 1.15. Change of the Normal Load Factor with Time during Disturbed Motion of an Aircraft After a Sharp Turn of the Elevator: 1. Beginning (Original) Value for a Normal Load Factor n_{orig}; 2. Terminal (Established) Value for a Normal Load Factor n_{est}; 3. Maximum Value for a Normal Load Factor with Overshoot n_{over}.

Fig. 1.16. Effect of the Operation of a Turbojet Engine on the Aircraft's Directional Stability: P - Thrust of the Engine; M_{y_p} - Disturbing Moment of Yaw; Z_p - Lateral Force Acting on the Air Intake.

During sideslip in an aircraft with a turbojet engine, there is a lateral force Zp acting on the inlet into the air intake (as is shown in Figure 1.16), because of a turn of the air inflow into the interior air-intake channel. In the case of air intake into the nose, this force is seen as a forward motion of the aircraft's center of gravity, which leads to the appearance of a disturbing moment of yaw M_{y_p}, increasing with an increase in the engine's thrust (or, more precisely, the volume of air sucked through the engine) and the angle of sideslip.

Elastic deformations of the fuselage structure also cause a decrease in the aircraft's directional stability, which decreases the angle of attack for the vertical empennage. This aeroelastic effect increases at a constant Mach number with a decrease in flight altitude. The decrease in the degree of the aircraft's static stability causes large sideslip angles, particularly during unilateral breakdown of engines in a multi-engine aircraft, which /34
in some cases can even lead to the aircraft's stalling.

1.2. The Effect of the Angle of Attack and Altitude on the Characteristics of Stability and Maneuverability

One of the characteristic features of a modern supersonic aircraft is the possibility for piloting it during take-off, landing, and maneuvering, particularly at subsonic speeds and high flight altitudes, at much greater angles of attack than is allowable in subsonic aircraft (Fig. 1.17). The latter is due mainly to an increase in the critical angles of attack, which permit expanding the range of operational angles of attack in supersonic aircraft, in comparison with subsonic aircraft. At the same time, the de-

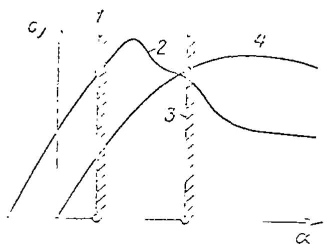


Fig. 1.17. Limits of the Ranges of Operational Angles of Attack in Subsonic and Supersonic Aircraft: 1. Limit of the Range of Operational Angles of Attack (Maximum Allowable Value in a Normal Flight Operation for an Aircraft) in a Subsonic Aircraft; 2. Function $c_y = f(\alpha)$ in a Subsonic Aircraft; 3. Limit of the Range of Operational Angles of Attack in a Supersonic Aircraft; 4. Function $c_y = f(\alpha)$ in a Supersonic Aircraft.

pendence of the characteristics for static stability and maneuverability on the angle of attack seems much more significant in supersonic aircraft than in subsonic ones. In this case, in the range of relatively small angles of attack, this dependence is usually much weaker than at average angles of attack, and even more so at large angles of attack. In the range of large subcritical angles of attack, there can be a highly non-linear dependence between these characteristics and the angle of attack. Such nonlinearity is seen most clearly in the longitudinal and directional (mainly lateral) stability of the aircraft.

(a) STABILITY WITH LOAD FACTOR

At subsonic and sonic flight speeds in some aircraft with sweptback wings, in the case of large angles of attack, there is an instability (so-called "dip") in the load factor accompanied by a tendency of the aircraft to dive spontaneously. This disadvantage acquires particularly important significance from the point of view of safety in piloting within the range of high subsonic (close to $M = 1$) flight speeds, at which there is usually the most marked appearance of instability in load factor (if this generally occurs in the given aircraft). At supersonic Mach numbers of flight, there is usually no instability in the load factor because the position for neutral centering is much closer to the rear. /35

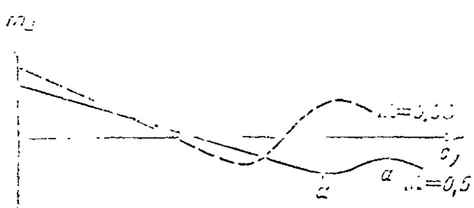


Fig. 1.18. Function $m_z = f(c_y)$ for an Aircraft with Sweptback Wings.

Typical relationships between the coefficient for the pitch moment m_z and the coefficient of lift for an aircraft with sweptback wings, for which there is stability with load factor in a certain range of angles of attack, are shown in Figure 1.18. The range of instability with load factor is characterized by the positive slope of the curve for $m_z = f(c_y)$: for example, in the curve segment *aa* which corresponds to $M = 0.5$.

The occurrence of instability with load factor in aircraft with straight sweptback wings is caused by the following factors:

(a) Shift forward of the aircraft's aerodynamic center as a result of a change in the circulation distribution (aerodynamic stress) for the wingspan, with an increase in the angle of attack;

(b) Change in the downwash field produced by the wing and the fuselage in the region of the horizontal empennage, with an increase in the angle of attack and a decrease in the reducing aerodynamic moment of pitch caused by the tail empennage;

(c) Increase in the disturbing component of the aerodynamic pitch moment, caused by the lift of the fuselage with an increase in the aircraft's angle of attack;

(d) Formation of a disturbing pitch moment by the turbojet engine, operating in an aircraft with air intakes in the nose.

For steady flow, the position of the aircraft's aerodynamic center does not depend on the value of the angle of attack. With the appearance of flow separation, the position of the center begins to achange. A straight sweptback wing usually has distribution of aerodynamic load over the wingspan such that its tips usually sustain more load than in a wing at right angles to the fuselage. Such a particular feature of the operation of a straight sweptback wing is due to the greater effective angles of attack in its wingtip sections, in comparison with the base of the wing. But this leads to the appearance of considerable pressure differentials along the wingspan, causing thickening (overflow) of the boundary layer in the direction of the wingtips.

The presence of larger angles of attack and a thickened boundary layer leads to the appearance of flow separation on the upper surface of the wing in these sections much earlier than in the middle sections. Therefore, with an increase in the angles of attack for the wing to those values at which there is flow separations at the wingtip sections (Fig. 1.19), there is a loss in lift at the wingtips ($-\Delta Y$), while the lift continues to increase in the middle with a further increase in the angle of attack. In this case, the aerodynamic center of the wing, and thus of the entire aircraft, moves forward. /36

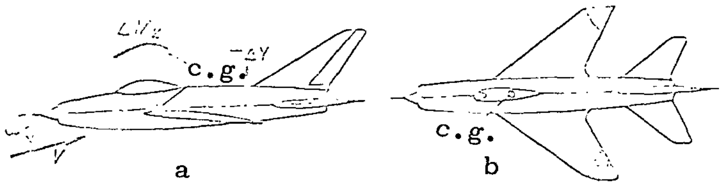


Fig. 1.19. Causes of the Occurrence of Instability in Load Factor Because of Wingtip Flow Separation on the Outer Surface of a Wing (the Areas of Flow Separation are Shaded).

Thus, when an aircraft with sweptback wings has no special means of countering the occurrence of terminal separation, the increase in the angle of attack leads to the appearance, and then to an intensive increase, of the regions for flow separation at the wingtips. When the angle of attack reaches values indicating the beginning of terminal flow separation from the wing, the degree of static stability in load factor for the aircraft begins to decrease, and then continues to fall with a further increase in the angle of attack. In this case, the more the pilot increases the aircraft's angle of attack, the larger the region for wingtip flow separation becomes, and the more the reducing aerodynamic moment of pitch decreases. In shifting the aerodynamic center so far forward that it is in front of the center of gravity, there arises a disturbing aerodynamic moment of pitch ΔM_Z which itself causes a further spontaneous increase in the angle of attack. The increase in the angle of attack then involves expansion of the wingtip flow separation region over the wingspan and a further increase in the moment ΔM_Z, etc. Longitudinal static instability with load factor then occurs. Instability with load factor can increase additionally by virtue of the aircraft's elastic structural deformations, and particularly because of the effect of the system for controlling the ailerons. The latter circumstance, in certain cases, can cause simultaneous upward movement ("floating up") of both ailerons, for which the non-uniform distribution of aerodynamic load over the wingspan increases at large angles of attack. Another factor which intensi- /37
fies load factor instability is the air intake into the nose. Such an air intake, when the angle of attack increases, increases the disturbing moment of pitch M_{zP} (the subscript "P" means that this moment depends on the thrust of the turbojet engine, which has the symbol P) produced by a normal force Y_P during operation of the turbojet engine (Fig. 1.20). With an increase in the engines thrust, this disturbing moment increases.

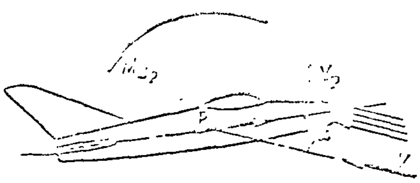


Fig. 1.20. Causes for Air Intake at Large Angles of Attack, for the Disturbing Moment M_{zP} during Operation of a Turbojet Engine.

Emergence at large angles of attack, for which there is instability with load factor, is not without danger, since the pilot cannot always succeed in countering further unintentional pitching of the aircraft.

(b) LATERAL STABILITY OF AN AIRCRAFT

The degrees of an aircraft's directional (m_y^β) and longitudinal (m_x^β) static stability change very greatly with the angle of attack.

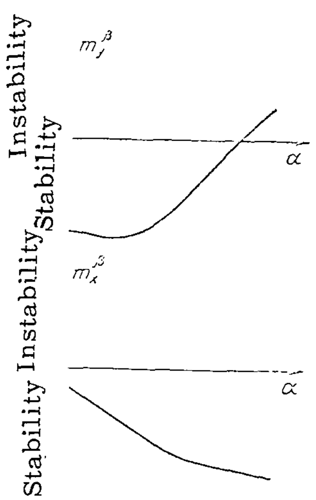


Fig. 1.21. Typical Change in the Degree of an Aircraft's Path m_y^β and Longitudinal m_x^β Static Stability as a Function of the Angle of Attack.

The directional static stability of a supersonic aircraft decreases significantly with an increase in the angle of attack (see Figures 1.1 and 1.21). This decrease is particularly strong at supersonic flight speeds. At high (often much less than critical) angles of attack, there can possibly even be directional instability for the aircraft. Longitudinal stability increases noticeably during an increase in the angle of attack (see Fig. 1.21). The decrease in the directional stability, and the increase in the longitudinal static stability are the cause of an increase in the lateral oscillations of the aircraft, of a sudden "drop" in roll during sideslip, and of an increase in the ratio of maximum angular velocities for roll and yaw, which is characterized by the index $\kappa = \frac{\omega_{x\,\max}}{\omega_{y\,\max}}$. An increase in the value for κ leads to the circumstance that even fairly low angular velocities of yaw cause high angular velocities (and thus large angles, also) of roll. This complicates the piloting of an aircraft with sweptback wings at large angles of attack. /38

(c) EFFECT OF THE FLIGHT ALTITUDE ON THE DYNAMIC CHARACTERISTICS OF AN AIRCRAFT

The effect of the flight altitude on the dynamic characteristics of an aircraft is very substantial. It is determined by three main factors: increase in the Mach number (during flight at a steady instrument speed with an increase in altitude, the Mach number increases), increase in the angle of attack (because of the decrease in the density of the air, the angle of attack necessary for rectilinear horizontal flight increases), and a decrease in the density of the air. Aerodynamic damping of the aircraft's vibrations (longitudinal and lateral) is greatly weakened when the Mach numbers increase and the density of the air decreases. (This, in particular, is substantiated by the example for the change in the degree of pitch damping $m_z^{\bar{\omega}_z}$ with Mach number, which is shown on the graph in Figure 1.12). As a result of the decrease in the aerodynamic damping moments, the attenuation of the aircraft's disturbed motion at high altitudes deteriorates (Fig. 1.22). During climb at a constant Mach number, as a result of a decrease in the rate of thrust, the aerodynamic reducing moments of yaw and pitch also decrease; this can also be caused by a decrease in static stability with an increase in the angle of attack. Therefore, the effect of inertial moments on an aircraft's dynamic characteristics increases at high altitudes.

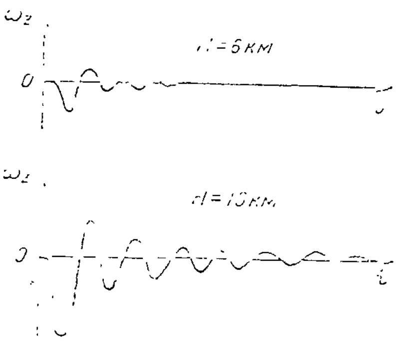


Fig. 1.22. Damping characteristics of Longitudinal Oscillations of an Aircraft, Consisting of Short-term Variations of the Rudder Position at High and Low Altitudes.

Because of the decrease in the damping of an aircraft's natural vibrations and the significant increase in the value of κ, the aircraft's "travel density" (as the pilots say) is worse at high altitudes. With an increase in altitude with a fixed instrument flight speed, maneuverability of the aircraft is also much worse, i.e., the aircraft's ability to "respond to the stick". The aircraft becomes sluggish to control, and the lag in its response to a pilot's action increases greatly. This lowers accuracy in piloting and complicates the pilot's work in the absence of a system for automatic damping and stabilization.

With an increase in Mach num- /39
bers and angle of attack at high altitudes, the significance of the features described earlier for a supersonic aircraft's stability and maneuverability, coupled with the effect of these parameters, is increased as well. The non-linearity of the dependence between the aerodynamic forces and moments and the angles of attack and sideslip is worse. At high altitudes in particular, the overshoot of load factor and the intensity of the occurrence of an interaction between the longitudinal and lateral motions of an aircraft during rotation increase significantly.

The features described in this chapter for the stability, maneuverability, and piloting characteristics complicate piloting in one way or another. Under certain conditions, they can cause dangerous situations in flight. However, they cannot in themselves be attributed to the factors which decrease supersonic aircraft flight safety during correct operation in flight. Dangerous situations in flight, especially the aircraft's entry into critical regimes, can occur because of these specific features only in extreme cases: when the rules for an aircraft's operation in flight are broken (a need for instruction in piloting), when there are defects in physical parts (especially in the control system and the engine), or when a pilot makes gross errors. During normal flight, the flight safety of modern supersonic aircraft is guaranteed in all operational flight regimes.

CHAPTER 2

EFFECT OF ROTATION OF AN AIRCRAFT ON THE ESSENTIAL CHARACTERISTICS OF ITS MANEUVERABILITY. CRITICAL ANGULAR VELOCITIES OF ROLL /40

2.1. Motion Equation of an Aircraft

(a) COMPLETE AND SIMPLIFIED EQUATIONS.

The motion equations of an aircraft viewed as a solid can be written (in the general case) in vectorial form as follows:

$$m\frac{d\vec{V}}{dt}+m[\vec{\Omega},\ \vec{V}]=\vec{R}_c+\vec{G}, \tag{2.1}$$

$$\frac{d\vec{L}}{dt}+[\vec{\Omega},\ \vec{L}]=\vec{M}; \tag{2.2}$$

where

$\vec{R}_c$ is the principal vector of all external forces acting on the aircraft (except the force of gravity).

$\vec{G}$ is the (weight) vector.

$\vec{M}$ is the principal vector of all external moments acting on the aircraft.

$\vec{L}$ is the vector of the resultant moment of the inertia of the aircraft.

$\vec{V}$ is the vector of the flight speed of the aircraft.

$\vec{\Omega}$ is the vector of the angular velocity of the aircraft's rotation.

m is the mass of the aircraft.

The system of vector equations (2.1) and (2.2) is a system of nonlinear differential equations with variable coefficients and are, as we know, not of interest in the general form. To obtain the partial solutions which are of interest to us, we must simplify the above system of equations. To do this, we must first employ a method of linearizing them and then write the linearized motion /41 equations (2.1) and (2.2) in projections of the forces along the axes of the system of coordinates associated with the aircraft. Furthermore, in order to simplify further the linearized motion equations of the aircraft obtained in this manner, in various cases we will disregard as much as possible the individual interactions of the parameters characterizing the motion of the aircraft.

Solutions of the linearized equations of the disturbed motion of an aircraft do not permit us to obtain precise quantitative data, since this necessarily involves a disregard of nonlinear interactions of various parameters of the aircraft's motion, which often play an important role (especially in critical flight regimes). However, the linearization method does permit us to obtain sufficiently clear solutions, which provide some idea of the qualitative aspect of the observed phenomena and a deeper understanding of their physical significance. They also make it possible to analyze the effect of various factors on the maneuverability and stability of the aircraft.

To form a system of linearized scalar equations for the motion of an aircraft, let us express the vectors of angular velocity of rotation of the aircraft $\vec{\Omega}$ and the flight speed $\vec{V}$ as the sums of two components: the constant value of the corresponding parameter in the original stable curvilinear motion of the aircraft and the additional increase (change) in this parameter in the course of the subsequent disturbed motion of the aircraft:

$$\vec{\Omega} = \vec{\omega}_0 + \vec{\omega}, \tag{2.3}$$

$$\vec{V} = \vec{V}_0 + \Delta\vec{V} \tag{2.4}$$

where

$\vec{\omega}_0$ is the vector of the stable angular velocity of the rotation of the aircraft in the initial flight regime;

$\vec{\omega}$ is the change in the vector of angular velocity of rotation of the aircraft in disturbed motion, with respect to its value in the initial flight regime.

$\vec{V}_0$; $\Delta\vec{V}$ are the speed vector of the initial stable flight regime and the change of the flight speed vector in disturbed motion with respect to its value in the original flight regime, respectively.

Similarly, let us represent the angles of attack and sideslip (α and β, respectively), as well as the coefficients of the lifting and side force aerodynamic forces of the aircraft (c_y and c_z, respectively) in the form of the following sums:

$$\alpha = \alpha_0 + \Delta\alpha,$$
$$\beta = \beta_0 + \Delta\beta,$$
$$c_y = c_{y_0} + \Delta c_y,$$
$$c_z = c_{z_0} + \Delta c_z.$$

For an approximate determination of the criteria of maneuverability and stability, we will consider the motion equations of the aircraft with the following assumptions: /42

(a) The effect of the aircraft's weight on the nature of its disturbed motion is negligibly small.

(b) The change in altitude during the time that the motion in question is going on is negligibly small, so that $\rho_H \approx$ const.

(c) Instability of flow over the aircraft during the disturbed motion in question will be disregarded.

(d) The sideslip angle $\beta_0 = 0$, so that $\beta = \Delta\beta$.

(e) The angles α and β are sufficiently small so that:

$$\cos\alpha = \cos(\alpha_0 + \Delta\alpha) = \cos\alpha_0 \cos\Delta\alpha - \sin\alpha_0 \sin\Delta\alpha \approx 1 - \alpha_0\Delta\alpha;$$
$$\sin\alpha = \sin(\alpha_0 + \Delta\alpha) = \sin\alpha_0 \cos\Delta\alpha + \cos\alpha_0 \sin\Delta\alpha \approx \alpha_0 + \Delta\alpha;$$
$$\cos\beta \approx 1;$$
$$\sin\beta \approx \beta;$$

(f) The angle between the principal axes of inertia of the aircraft and its related axes are sufficiently small so that the related system of coordinates can be assumed to coincide with its principal axes of inertia.

(g) The degree of change in all parameters of the disturbed motion of the aircraft is sufficiently small so that their squares and derivatives can be disregarded as being values which are of much smaller size, i.e., we will use the method of small perturbations.

In view of the above assumptions, the projections of the flight speed vector and its derivatives, as well as the projections of the vector of angular velocity of rotation of the aircraft on the axes of the system of coordinates related to it can be represented in the following form (here and in the following, the time derivatives will be marked with a dot above the differential variable):

$$V_x = V\cos\alpha\cos\beta \approx (V_0+\Delta V)(1-\alpha_0\Delta\alpha) \approx V_0+\Delta V - V_0\alpha_0\Delta\alpha;$$
$$V_y = -V\sin\alpha\cos\beta \approx (V_0+\Delta V)(-\alpha_0-\Delta\alpha) \approx -V_0\alpha_0 - V_0\Delta\alpha - \Delta V\,\alpha_0;$$
$$V_z = V\sin\beta \approx (V_0+\Delta V)\beta \approx V_0\beta;$$
$$\dot V_x \approx \Delta\dot V - V_0\alpha_0\Delta\dot\alpha;$$
$$\dot V_y \approx -V_0\Delta\dot\alpha - \alpha_0\Delta\dot V;$$
$$\dot V_z \approx V_0\dot\beta;$$
$$\Omega_x = \omega_{x_0}+\omega_x; \quad \Omega_y = \omega_{y_0}+\omega_y; \quad \Omega_z = \omega_{z_0}+\omega_z.$$

By projecting the vectors which enter into (2.1) and (2.2) along the axes of the system of coordinates which we are using, and keeping in mind the assumptions made above, we obtain the following system of six scalar equations of motion for the aircraft: /43

$$m[\Delta\dot V - V_0\alpha_0\Delta\dot\alpha + V_0\beta(\omega_{y_0}+\omega_y) + (V_0\alpha_0 + V_0\Delta\alpha + \alpha_0\Delta V)(\omega_{z_0}+\omega_z)] = \sum X; \tag{2.5}$$

$$m[-V_0\Delta\dot\alpha - \alpha_0\Delta\dot V + (V_0+\Delta V - V_0\alpha_0\Delta\alpha)(\omega_{z_0}+\omega_z) - V_0\beta(\omega_{x_0}+\omega_x)] = \sum Y; \tag{2.6}$$

$$m[V_0\dot\beta - (V_0\alpha_0 + V_0\Delta\alpha + \alpha_0\Delta V)(\omega_{x_0}+\omega_x) - (V_0+\Delta V - V_0\alpha_0\Delta\alpha)(\omega_{y_0}+\omega_y)] = \sum Z; \tag{2.7}$$

$$J_x\dot\omega_x + (J_z - J_y)(\omega_{y_0}+\omega_y)(\omega_{z_0}+\omega_z) = \sum M_x; \tag{2.8}$$

$$J_y\dot\omega_y + (J_x - J_z)(\omega_{x_0}+\omega_x)(\omega_{z_0}+\omega_z) = \sum M_y; \tag{2.9}$$

$$J_z\dot\omega_z + (J_y - J_x)(\omega_{x_0}+\omega_x)(\omega_{y_0}+\omega_y) = \sum M_z. \tag{2.10}$$

The projections of all external forces and moments acting on the aircraft along the axes of the system of coordinates, which enter into the system of equations (2.5) to (2.10), can be represented as follows:

$$\sum X = X_0 + X^{c_y}\Delta c_y + X^V\Delta V; \tag{2.11}$$

$$\sum Y = Y_0 + Y^{\alpha}\Delta\alpha + Y^{V}\Delta V; \qquad (2.12)$$

$$\sum Z = Z_0 + Z^{c_y}\Delta c_y + Z^{\beta}\beta + Z^{V}\Delta V; \qquad (2.13)$$

$$\sum M_x = M_{x_0} + M_x^{c_y}\Delta c_y + M_x^{\beta}\beta + M_x^{\omega_x}\omega_x + M_x^{\omega_y}\omega_y + \qquad (2.14)$$

$$+ M_x^{\dot\beta}\dot\beta + M_x^{V}\Delta V + M_x^{\delta_a}\Delta\delta_a;$$

$$\sum M_y = M_{y_0} + M_y^{c_y}\Delta c_y + M_y^{\beta}\beta + M_y^{\omega_y}\omega_y + M_y^{\omega_x}\omega_x +$$

$$+ M_y^{\dot\beta}\dot\beta + M_y^{V}\Delta V + M_y^{\delta_n}\Delta\delta_n - J_p\omega_p(\omega_{z_0} + \omega_z); \qquad (2.15)$$

$$\sum M_z = M_{z_0} + M_z^{c_y}\Delta c_y + M_z^{\beta}\beta + M_z^{\omega_z}\omega_z + M_z^{\dot\alpha}\Delta\dot\alpha +$$

$$+ M_z^{V}\Delta V + M_z^{\delta_B}\Delta\delta_B + J_p\omega_p(\omega_{y_0} + \omega_y). \qquad (2.16)$$

Here Δ is a symbol representing the increase of a parameter. The subscripts represent the derivatives of the aerodynamic forces and moments for the corresponding variable. Within the range of variations in the motion parameters of the aircraft α, V, and β in the course of its disturbed motion with which we are concerned, the engine thrust will be assumed constant. Therefore, the engine thrust is not mentioned as such in (2.11) to (2.16), but its components are included in the values X_0, Y_0, and M_{z0}. /44

The equations for the initial stable flight regime are

$$mV_0\alpha_0\omega_{z_0} = X_0;$$
$$mV_0\omega_{z_0} = Y_0;$$
$$-mV_0(\alpha_0\omega_{x_0} + \omega_{y_0}) = Z_0;$$
$$(J_z - J_y)\omega_{y_0}\omega_{z_0} = M_{x_0};$$
$$(J_x - J_z)\omega_{x_0}\omega_{z_0} = M_{y_0} - J_p\omega_p\omega_{z_0};$$
$$(J_y - J_x)\omega_{x_0}\omega_{y_0} = M_{z_0} + J_p\omega_p\omega_{y_0}.$$

If we take into account the motion equations as written here for the initial flight regime, (2.5) to (2.10) can be rewritten as follows:

$$\frac{\Delta\dot{V}}{V_0} - \alpha_0\Delta\dot{\alpha} + \omega_{y_0}\beta + \alpha_0\omega_z + \omega_{z_0}\Delta\alpha -$$
$$- \frac{\omega_{z_0}\alpha_0}{V_0}\Delta V = \frac{X^{c_y}}{mV_0}\Delta c_y + \frac{X^V}{mV_0}\Delta V; \tag{2.17}$$

$$-\Delta\dot{\alpha} - \frac{\alpha_0}{V_0}\Delta\dot{V} + \frac{\omega_{z_0}}{V_0}\Delta V - \omega_{z_0}\alpha_0\Delta\alpha + \omega_z - \omega_{x_0}\beta =$$
$$= \frac{Y^\alpha}{mV_0}\Delta\alpha + \frac{Y^V}{mV_0}\Delta V; \tag{2.18}$$

$$\dot{\beta} - \omega_{x_0}\Delta\alpha - \frac{\omega_{x_0}\alpha_0}{V_0}\Delta V - \alpha_0\omega_x - \frac{\omega_{y_0}}{V_0}\Delta V + \omega_{y_0}\alpha_0\Delta\alpha - \omega_y =$$
$$= \frac{Z^{c_y}}{mV_0}\Delta c_y + \frac{Z^\beta}{mV_0}\beta + \frac{Z^V}{mV_0}\Delta V; \tag{2.19}$$

$$J_x\dot{\omega}_x + (J_z - J_y)(\omega_{y_0}\omega_z + \omega_{z_0}\omega_y) = M_x^{c_y}\Delta c_y + M_x^\beta\beta + M_x^{\omega_x}\omega_x +$$
$$+ M_x^{\omega_y}\omega_y + M_x^{\dot{\beta}}\dot{\beta} + M_x^V\Delta V + M_x^{\delta_a}\Delta\delta_a; \tag{2.20}$$

$$J_y\dot{\omega}_y + (J_x - J_z)(\omega_{x_0}\omega_z + \omega_{z_0}\omega_x) = M_y^{c_y}\Delta c_y + M_y^\beta\beta +$$
$$+ M_y^{\omega_y}\omega_y + M_y^{\omega_x}\omega_x + M_y^{\dot{\beta}}\dot{\beta} + M_y^V\Delta V + M_y^{\delta_H}\Delta\delta_H - J_p\omega_p\omega_z; \tag{2.21}$$

$$J_z\dot{\omega}_z + (J_y - J_x)(\omega_{x_0}\omega_y + \omega_{y_0}\omega_x) = M_z^{c_y}\Delta c_y + M_z^\beta\beta +$$
$$+ M_z^{\omega_z}\omega_z + M_z^{\dot{\alpha}}\Delta\dot{\alpha} + M_z^V\Delta V + M_z^{\delta_B}\Delta\delta_B + J_p\omega_p\omega_y. \tag{2.22}$$

In addition, the derivatives of the aerodynamic forces and moments can be replaced by the corresponding derivatives of their coefficients, using the following equations for this purpose:

$$\left.\begin{aligned}
X^{c_y} &= c_x^{c_y} S \frac{\varrho_H V_0^2}{2}; & X^V &= c_x^M S M_0 \frac{\varrho_H V_0}{2};\\
Y^\alpha &= c_y^\alpha S \frac{\varrho_H V_0^2}{2}; & Y^V &= c_y^M S M_0 \frac{\varrho_H V_0}{2};\\
Z^{c_y} &= c_z^{c_y} S \frac{\varrho_H V_0^2}{2}; & Z^\beta &= c_z^\beta S \frac{\varrho_H V_0^2}{2};\\
Z^V &= c_z^M S M_0 \frac{\varrho_H V_0}{2}; & M_x^{c_y} &= m_x^{c_y} S l \frac{\varrho_H V_0^2}{2};
\end{aligned}\right\}$$

$$
\left.
\begin{aligned}
&M_x^{\beta}=m_x^{\beta}Sl\,\frac{\varrho_H V_0^2}{2}; && M_x^{\omega_x}=m_x^{\bar{\omega}_x}Sl^2\frac{\varrho_H V_0}{4};\\
&M_x^{\omega_y}=m_x^{\bar{\omega}_y}Sl^2\frac{\varrho_H V_0}{4}; && M_x^{\delta}=m_x^{\delta}Sl\,\frac{\varrho_H V_0^2}{2};\\
&M_x^{V}=m_x^{M}SlM_0\,\frac{\varrho_H V_0}{2}; && M_x^{\delta_a}=m_x^{\delta_a}Sl\,\frac{\varrho_H V_0}{2};\\
&M_y^{c_y}=m_y^{c_y}\,Sl\,\frac{\varrho_H V_0^2}{2}; && M_y^{\beta}=m_y^{\beta}Sl\,\frac{\varrho_H V_0^2}{2};\\
&M_y^{\omega_y}=m_y^{\bar{\omega}_y}Sl^2\frac{\varrho_H V_0}{4}; && M_y^{\omega_x}=m_y^{\bar{\omega}_x}Sl^2\frac{\varrho_H V_0}{4};\\
&M_y^{\delta}=m_y^{\delta}Sl\,\frac{\varrho_H V_0^2}{2}; && M_y^{V}=m_y^{M}SlM_0\frac{\varrho_H V_0}{2};\\
&M_y^{\delta_н}=m_y^{\delta_н}Sl\,\frac{\varrho_H V_0^2}{2}; && M_z^{c_y}=m_z^{c_y}Sb_A\,\frac{\varrho_H V_0^2}{2};\\
&M_z^{\delta}=m_z^{\delta}Sb_A\,\frac{\varrho_H V_0^2}{2}; && M_z^{\omega_z}=m_z^{\bar{\omega}_z}Sb_A^2\frac{\varrho_H V_0}{2};\\
&M_z^{\dot{\alpha}}=m_z^{\dot{\alpha}}Sb_A\,\frac{\varrho_H V_0^2}{2}; && M_z^{V}=m_z^{M}Sb_AM_0\frac{\varrho_H V_0}{2};\\
&M_z^{\delta_B}=m_z^{\delta_B}Sb_A\,\frac{\varrho_H V_0^2}{2}.
\end{aligned}
\right\}
\qquad (2.23)
$$

(b) CHANGES IN THE COEFFICIENTS OF THE AERODYNAMIC FORCES AND THE ANGLES OF ATTACK AND SIDESLIP

/46

In determining the criteria for maneuverability of an aircraft, it will be necessary to replace the changes Δc_y, Δc_z, $\Delta\alpha$ and β (changes in these parameters relative to their initial values) in the motion equations, by the changes Δn_y, ΔV, and Δn_z. Let us begin by finding the relationship between the change in the coefficient of lift Δc_y and the changes in the flight speed ΔV and the vertical load factor Δn_y. To do this, let us represent the vertical load factor as follows:

$$
\begin{aligned}
n_y = n_{y_0} + \Delta n_y &= \frac{(c_{y_0}+\Delta c_y)\,S\varrho_H\,(V_0+\Delta V)^2}{2G} \approx \\
&\approx n_{y_0} + \frac{n_{y_0}}{c_{y_0}}\,\Delta c_y + \frac{2n_{y_0}}{V_0}\,\Delta V.
\end{aligned}
\qquad (2.24)
$$

From this we obtain the expression for the change in the coefficient of lift with a change in the flight regime:

$$
\Delta c_y = \frac{c_{y_0}}{n_{y_0}}\left(\Delta n_y - \frac{2n_{y_0}}{V_0}\,\Delta V\right) \qquad (2.25)
$$

Now let us express the change in the angle of attack $\Delta\alpha$ by the changes Δn_y and ΔV. The coefficient of lift of the aircraft will be considered to be a function of two variables: $c_y = c_y\,(\alpha, M)$. Then

$$\frac{dc_y}{d\alpha}=\frac{\partial c_y}{\partial \alpha}+\frac{\partial c_y}{\partial M}\frac{dM}{d\alpha}=c_y^{\alpha}+c_y^{M}\frac{dM}{dc_y}\frac{dc_y}{d\alpha}.$$

From this we obtain

$$\frac{dc_y}{d\alpha}=\frac{c_y^{\alpha}}{1+c_y^{M}\frac{dM}{dc_y}}\approx\frac{c_y^{\alpha}}{1-c_y^{M}\frac{\Delta M}{\Delta c_y}}. \tag{2.26}$$

The change in the angle of attack of the aircraft is expressed by:

$$\Delta\alpha=\frac{\Delta c_y}{\frac{dc_y}{d\alpha}}.$$

Now let us substitute the values Δc_y from (2.25) and $dc_y/d\alpha$ from (2.26), and replace ΔM by $\Delta V/a = (M_0/V_0)\Delta V$, where a is the speed of sound at the flight altitude in question; we then have: /47

$$\Delta\alpha=A_1\Delta n_y+A_2\Delta V. \tag{2.27}$$

Here:

$$A_1=\frac{c_{y_0}}{n_{y_0}c_y^{\alpha}} \tag{2.28}$$

and

$$A_2=-\frac{1}{c_y^{\alpha}}\left(\frac{2c_{y_0}}{V_0}+\frac{M_0}{V_0}c_y^{M}\right). \tag{2.29}$$

Similarly, we can also find the relationship of a change in the glide angle β to changes in the parameters of the flight trajectory which are perceived directly by the pilot (n_z, n_y, and V).

For this purpose, we can use the following formula for the side force load factor n_z:

$$n_z=n_{z_0}+\Delta n_z=\frac{(c_{z_0}+\Delta c_z)\,S\varrho_H\,(V_0+\Delta V)^2}{2G}\approx$$
$$\approx n_{z_0}+\frac{n_{z_0}}{c_{z_0}}\Delta c_z+\frac{2n_{z_0}}{V_0}\Delta V. \tag{2.30}$$

From this we obtain:

$$\Delta c_z = \frac{c_{z_0}}{n_{z_0}}\left(\Delta n_z - \frac{2n_{z_0}}{V_0}\Delta V\right). \tag{2.31}$$

In the special case when $n_{z_0} = c_{z_0} = 0$, and $n_{y_0} = 1$, we obtain the following from the left-hand side of (2.30):

$$\Delta c_z = c_{y_0}\Delta n_z. \tag{2.32}$$

The side force coefficient will be considered to be a function of three variables: $c_z = c_z(\beta, c_y, M)$. The total derivative $dc_z/d\beta$ in this case will be equal to

$$\frac{dc_z}{d\beta} = \frac{\partial c_z}{\partial \beta} + \frac{\partial c_z}{\partial c_y}\frac{dc_y}{d\beta} + \frac{\partial c_z}{\partial M}\frac{dM}{d\beta} =$$
$$= c_z^\beta + c_z^{c_y}\frac{dc_y}{dc_z}\frac{dc_z}{d\beta} + c_z^M\frac{dM}{dc_z}\frac{dc_z}{d\beta}.$$

From this we obtain

$$\frac{dc_z}{d\beta} = \frac{c_z^\beta}{1 - c_z^{c_y}\frac{dc_y}{dc_z} - c_z^M\frac{dM}{dc_z}} \approx \frac{c_z^\beta}{1 - c_z^{c_y}\frac{\Delta c_y}{\Delta c_z} - c_z^M\frac{\Delta M}{\Delta c_z}}. \tag{2.33}$$

Let us express β in the form /48

$$\beta = \frac{\Delta c_z}{\frac{dc_z}{d\beta}}.$$

By substituting into this equation the values Δc_z from (2.31), Δc_y from (2.25), and $dc_z/d\beta$ from (2.33), we will have:

$$\beta = B_1\Delta n_z + B_2\Delta n_y + B_3\Delta V, \tag{2.34}$$

where

$$B_1 = \frac{c_{z_0}}{n_{z_0}c_z^\beta}, \tag{2.35}$$

$$B_2 = -\frac{c_{y_0}c_z^{c_y}}{n_{y_0}c_z^\beta} \tag{2.36}$$

and

$$B_3 = \frac{1}{V_0 c_z^\beta}(2c_{y_0}c_z^{c_y} - M_0 c_z^M - 2c_{z_0}). \tag{2.37}$$

For the partial case $n_{z0} = c_{z0} = 0$ and $n_{y0} = 1$, we have:

$$\beta = b_1 \Delta n_z + b_2 \Delta n_y + b_3 \Delta V, \tag{2.38}$$

where

$$b_1 = \frac{c_{y0}}{c_z^{\beta}}, \tag{2.39}$$

$$b_2 = -\frac{c_{y0} c_z^{c_y}}{c_z^{\beta}}, \tag{2.40}$$

and

$$b_3 = \frac{1}{V_0 c_z^{\beta}} (2c_{y0} c_z^{c_y} - M_0 c_z^{M}). \tag{2.41}$$

2.2. Elevator Motion per Unit Load Factor

The first chapter dealt with the physical aspects of the interaction of the longitudinal and lateral motions of an aircraft while rotating. It was shown that with a certain combination of the parameters of motion, inertial, and aerodynamic characteristics of the aircraft, the above-mentioned interaction can lead to a loss of stability. Let us now consider the effect of rotation of the aircraft on the basic characteristics of its controllability.

Let us determine the displacement of the elevator (or movable stabilizer) per unit load factor during flight when the aircraft is rotating, and the initial flight regime is a quasistable motion of the aircraft with rolling: /49

$$\omega_x = \text{const} \neq 0; \quad \omega_{y0} = \omega_{z0} = 0; \quad n_{y0} = 1 \text{ and } n_{z0} = 0.$$

For the present, we will consider the side force coefficient of the aircraft to be the function of only two variables (for the sake of approximation): the angle of sideslip and the Mach number of flight, i.e., we will initially assume that $c_z(c_y) = 0$.

(a) DERIVATION OF THE EXPRESSION FOR THE MOTION OF THE ELEVATOR PER UNIT LOAD FACTOR

Under the assumptions listed above, the increase in the angle of the elevator or movable stabilizer required for a given change of flight parameters can be found from (2.22) as follows:

$$\Delta\delta_{в} = -\frac{1}{M_z^{\delta_{в}}} [J_z \dot{\omega}_z + (J_y - J_x) \omega_{x0} \omega_y - J_{y} \omega_{y} \omega_y - M_z^{c_y} \Delta c_y - M_z^{\beta} \beta - M_z^{\omega_z} \omega_z - M_z^{\dot{\alpha}} \Delta \dot{\alpha} - M_z^{V} \Delta V]. \tag{2.42}$$

The changes in the angular velocities of pitch and yaw (ω_z and ω_y, respectively) are likewise obtained from (2.18) and (2.19):

$$\omega_z = \frac{Y^{\alpha}}{mV_0}\Delta\alpha + \frac{Y^V}{mV_0}\Delta V + \Delta\dot{\alpha} + \frac{\alpha_0}{V_0}\Delta\dot{V} + \omega_{x_0}\beta; \tag{2.43}$$

$$\omega_y = -\frac{Z^{\beta}}{mV_0}\beta - \frac{Z^V}{mV_0}\Delta V + \dot{\beta} - \omega_{x_0}\Delta\alpha - \frac{\omega_{x_0}\alpha_0}{V_0}\Delta V - \alpha_0\omega_x. \tag{2.44}$$

The total angular velocity of roll Ω_x can be represented approximately by the expression

$$\Omega_x \approx \frac{d\gamma}{dt} = \frac{d\gamma}{d\beta}\frac{d\beta}{dt},$$

where $d\gamma/d\beta$ is independent of Δn_y, Δn_x and ΔV.

Consequently,

$$\omega_x = \frac{d\gamma}{d\beta}\dot{\beta} - \omega_{x_0}. \tag{2.45}$$

If we substitute the values ω_z, ω_y and ω_x from (2.43), (2.44) and (2.45) into (2.42) and then replace the values Δc_y, $\Delta\alpha$, and β in accordance with (2.25), (2.27) and (2.38), we will have a formula for determining $\Delta\delta_B$ from the changes in the parameters of the flight trajectory Δn_y, Δn_z and ΔV and their time derivatives:

$$\begin{aligned}\Delta\delta_B = {} & \delta_B^{n_y}\Delta n_y + \delta_B^{\dot{n}_y}\Delta\dot{n}_y + \delta_B^{\ddot{n}_y}\Delta\ddot{n}_y + \delta_B^{n_z}\Delta n_z + \\ & + \delta_B^{\dot{n}_z}\Delta\dot{n}_z + \delta_B^{\ddot{n}_z}\Delta\ddot{n}_z + \delta_B^{V}\Delta V + \delta_B^{\dot{V}}\Delta\dot{V} + \delta_B^{\ddot{V}}\Delta\ddot{V}.\end{aligned} \tag{2.46}$$

If we assume that the derivatives of the changes in the trajectory parameters with time are equal to zero and we differentiate (2.46) for vertical load factor, we will have the total motion of the elevator required for the aircraft to make the transition from an initial stable flight regime with $n_{y_0} = 1$, $n_{z_0} = 0$ and $V_0 = \text{const}$, to the also stable regime of curvilinear flight in space in the general case, with $n_y \neq 1$, $n_z \neq 0$, and $V \neq V_0$. This total motion of the elevator is represented by the total derivative /50

$$\frac{d\delta_B}{dn_y} = \frac{\partial\delta_B}{\partial n_y} + \frac{\partial\delta_B}{\partial n_z}\frac{dn_z}{dn_y} + \frac{\partial\delta_B}{\partial V}\frac{dV}{dn_y} =$$

$$= \delta_B^{n_y} + \delta_B^{n_z}\frac{dn_z}{dn_y} + \delta_B^{V}\frac{dV}{dn_y}.$$

Shifting from the differentials to the final increment, we have

$$\frac{d\delta_B}{dn_y} \approx \frac{\Delta\delta_B}{\Delta n_y} = \delta_B^n + \delta_B^z \frac{\Delta n_z}{\Delta n_y} + \delta_B^V \frac{\Delta V}{\Delta n_y}. \tag{2.47}$$

We find the expanded expressions for determining the partial derivatives involves here, δ_B^n, δ_B^z, and δ_B^V.

Using the calculations produced in determining (2.46), we have the following expression for δ_B^n.

$$\delta_B^n = \frac{1}{M_z^{\delta_B}} \left[-M_z^{c_y} \Delta c_y - M_z^{\omega_z} \frac{Y^c}{mV_0} A_1 + J_p \omega_p A_1 \omega_{x_0} - \right. \\ \left. - (J_y - J_x) A_1 \omega_{x_0}^2 \right]. \tag{2.48}$$

From the above it is clear that the value δ_B^n can be represented in the form

$$\delta_B^n = \delta_{B_0}^n + \delta_{B_1}^n \omega_{x_0} + \delta_{B_2}^n \omega_{x_0}^2. \tag{2.49}$$

The coefficients δ_{B1}^n and δ_{B2}^n in (2.49), which express the relationship between the value of δ_B^n and the angular velocity of roll, reflect the influence of the operating regime of the engine (coefficient δ_{B1}^n, including the parameters J_p and ω_p) and the nature of the mass distribution over the aircraft (the coefficient δ_{B2}^n the difference between the values J_x and J_y).

Having used the relationships (2.23) to express the derivatives of the aerodynamic forces and moments through the corresponding derivatives of the aerodynamic coefficients, we obtain the formulas for determining the coefficients of (2.49) as follows: /51

$$\delta_{B_0}^n = -\frac{c_{y_0} \sigma_n}{m_z^{\delta_B}}, \tag{2.50}$$

where

$\sigma_n = m_z^{c_y} + \frac{m_z^{\bar{\omega}_z}}{\mu_l}$ — is the static margin of longitudinal stability of the aircraft under normal load factor;

$\mu_l = \frac{2m}{\rho_H S b_A}$ — is the relative rate of the aircraft in longitudinal motion;

$$\delta^n_{B1} = \frac{J_p \omega_p \mu l c_{y0}}{K c_y^\alpha m_z^{\delta_B}}. \qquad (2.51)$$

Here $K = mV_0^2$ is the doubled kinetic energy of the aircraft. Finally,

$$\delta^n_{B2} = \frac{(J_x - J_y)\mu l c_{y0}}{K c_y^\alpha m_z^{\delta_B}}. \qquad (2.52)$$

(b) ANALYSIS OF THE EFFECT OF VARIOUS FACTORS ON THE MOTION OF THE ELEVATOR UNDER LOAD FACTOR

It is clear from (2.50) that with $\sigma_n < 0$ and $m_z^{\delta_B} < 0$, the coefficient δ^n_{B0} is negative. If $\omega_p < 0$ (left-hand rotation of the engine rotor, which also occurs as a rule in modern turbojet aircraft), then [as we see from (2.51)] the value of δ^n_{B1} will be positive. The greater the polar moment of inertia of the engine rotor and the higher the engine rpm, the greater will be the value of δ^n_{B1}, i.e., the greater will be the effect of the angular velocity of roll ω_{x0} on the derivative δ^n_B.

It follows from (2.52) that the value δ^n_{B2} is also positive when $J_y > J_x$ and $m_z^{\delta_B} < 0$, which also is the case as a rule in modern supersonic aircraft. In particular, it follows from this that the absolute value of the partial derivative decreases with an increase in the mass distribution along the longitudinal axis of the fuselage (with an increase in J_y) and the coefficient of lift c_{y0} (flight at high altitude). This decrease in the derivative δ^n_B is more pronounced as the value of ω_{x0} increases (according to the law of the quadratic parabola).

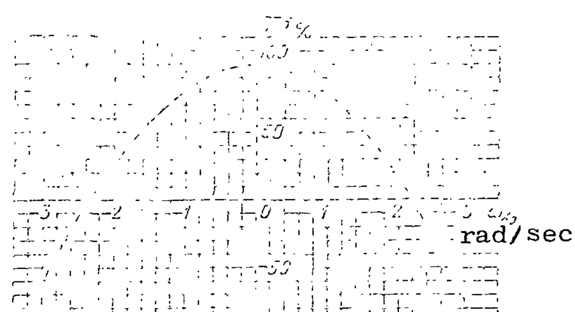


Fig 2.1: Example of the Dependence of $\overline{\delta^n_B}$ on the Value of the Initial Angular Velocity of Roll.

Figure 2.1 shows an example of the change in the relative value of the partial derivative $\overline{\delta^n_B}$ as a function of the value of the angular velocity of roll ω_{x0}. The value of $\overline{\delta^n_B}$ is expressed by the equation /52

$$\overline{\delta^n_B} = \frac{(\delta^n_B) \text{ for } \omega_{x0} \neq 0}{(\delta^n_B) \text{ for } \omega_{x0} = 0}\, 100\% = \frac{\delta^n_B}{\delta^n_{B0}}\, 100\%.$$

An example of the change in the relationship of the absolute values of the individual components in Equation (2.49) as a function of ω_{x0} is shown in Figure 2.2. Here the following legends are used:

$$\bar{\delta}^n_{B_0} = \frac{|\delta^n_{B_0}|}{\sum |\delta^n_{B_i} \omega^i_{x_0}|} 100\%;$$

$$\bar{\delta}^n_{B_1} = \frac{|\delta^n_{B_1} \omega_{x_0}|}{\sum |\delta^n_{B_i} \omega^i_{x_0}|} 100\%;$$

$$\bar{\delta}^n_{B_2} = \frac{|\delta^n_{B_2} \omega^2_{x_0}|}{\sum |\delta^n_{B_i} \omega^i_{x_0}|} 100\%,$$

where

$$\sum_{i=0}^{2} |\delta^n_{B_i} \omega^i_{x_0}| = |\delta^n_{B_0}| + |\delta^n_{B_1} \omega_{x_0}| + |\delta^n_{B_2} \omega^2_{x_0}|.$$

The initial parameters used in the calculations in this chapter are the following: /53

$G = 10\,000$ kg	$V_0 = 270$ m/sec	$c_y^{\bar{\omega}_z} = 0.05$
$m = 1020$ $\frac{\text{kg}\cdot\text{sec}^2}{\text{m}}$	$M_0 = 0.8$	$c_z^{\beta} = -0.8$
$S = 25$ m^2	$n_{y_0} = 1$	$c_z^{\bar{\omega}_x} = -0.01$
$b = 8$ m	$n_{z_0} = 0$	$m_z^{\alpha} = -0.09$
$l = 4$ m	$c_{y_0} = 0.26$	$m_z^{\dot{\alpha}} = -0.07$
$J_x = 500$ kg·m·sec^2	$\alpha_0 = 4.78°$	$m_z^{\bar{\omega}_z} = -2.8$
$J_y = 7500$ kg·m·sec^2	$\beta_0 = 0$	$m_z^{\beta} = -0.06$
$J_z = 7000$ kg·m·sec^2	$\varphi_1 = 45°$	$m_z^{\bar{\omega}_x} = 0.03$
$J_p = 8$ kg·m·sec^2	$\varphi_2 = 2.2$	$m_y^{\bar{\omega}_x} = -0.06$
$\omega_p = -400$ rad/sec	$J = 1\,102\,000$ kg·m·sec	$m_y^{\beta} = -0.18$
$H_0 = 10$ km	$c_y^{\alpha} = 3.15$	$m_y^{\bar{\omega}_y} = -0.39$

It is clear from the diagram in Figure 2.2. that in this particular example the relative value of the term with δ^n_{B1} in Equation (2.49), characterizing the effect of the gyroscopic moment of the engine rotor, is small and remains practically the same with an increase in the angular velocity of roll ω_{x0}. The relative value of the term δ^n_{B0} decreases with increasing ω_{x0}, and at ω_{x0} = 2 rad/sec it already amounts to about 50% of its initial value. In other words, the margin of longitudinal stability of the aircraft under stress is of critical influence on the value δ^n_B at low angular velocities of roll, but this effect decreases markedly when ω_{x0}

increases. The effect of the weight distribution over the aircraft upon the value δ_B^n is characterized by the term with $\delta_{B_2}^n$. It is clear from the diagram that at angular velocities of roll less than 1 rad/sec, this effect is relatively slight, while at high angular velocities of roll it is really quite significant. /54

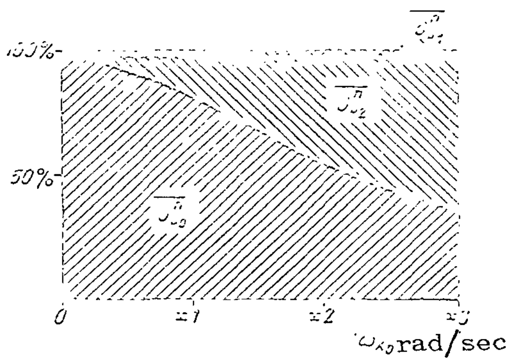


Fig. 2.2: Relationship of the Absolute Values of the Individual Components in the Expression for δ_B^n.

In a more complex instance the formula for the expression δ_B^n also becomes more complex.[1] Thus, for example, with $c_z^{c_y} \neq 0$, $n_{y_0} \neq 1$, $\omega_{x_0} \neq 0$ and $\omega_{y_0} \neq 0$, instead of (2.49) we will have

$$\delta_B^n = \delta_0^n + \delta_1^n \omega_{x_0} + \delta_2^n \omega_{x_0}^2 + \delta_3^n \omega_{y_0} + \delta_4^n \omega_{x_0} \omega_{y_0} + \delta_5^n \omega_{y_0}^2 + \delta_6^n \omega_{x_0}^2 \omega_{y_0} + \delta_7^n \omega_{x_0} \omega_{y_0}^2,$$

where, in particular,

$$\delta_0^n = \frac{\sigma_n}{n_y m_z^{c_y}} \frac{c_z^{c_y} m_z^{\beta}}{c_z^{\beta}} - \sigma_n - \frac{2J_x [illegible]}{J m_z^{[illegible]}} \left(m_x^{c_y} - \frac{c_z^{c_y} m_x^{\beta}}{c_z^{\beta}} \right)$$

and

$$\delta_7^n = \frac{2(J_x - J)(J_z - J_y)\rho_0^2 c_z c_z^{c_y}}{J [illegible]}.$$

Here, $J = mb_A V_0$ is the moment of the forward inertia of the aircraft.

For the sake of simplicity, the value J can be interpreted as the modulus of the inertial vector of the aircraft's center of gravity relative to a point whose distance from the center of gravity is equal to b_A and lies on the axis oy, perpendicular to the initial speed vector $\vec{V}_0$.

By analogy with the method by which (2.49) was obtained to determine δ_B^n, we can also find the expression for the partial

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[1] For further details, see: Kotik, M.G.: Upravlyayemost' sverkhzvukovykh samoletov v krivolineynom polete (Maneuverability of Supersonic Aircraft in Curvilinear Flight). Trudy VVIOLKA imeni Prof. N.E. Zhukovskiy, No. 1027, 1964.

derivative δ_B^z:

where

$$\delta_B^z = \delta_{B_0}^z + \delta_{B_1}^z \omega_{x_0}, \tag{2.53}$$

$$\delta_{B_0}^z = \frac{c_{y_0}}{m_z^{\delta_B}}\left(\frac{J_p \omega_p}{J} - \frac{m_z^\beta}{c_z^\beta}\right), \tag{2.54}$$

and

$$\delta_{B_1}^z = \frac{c_{y_0}}{m_z^{\delta_B}}\left(\frac{J_x - J_y}{J} - \frac{b_A c_{y_0} m_z^{\bar{\omega}_z}}{Y_0 c_z^\beta}\right). \tag{2.55}$$

The coefficient δ_B^z is always positive when $J_y > J_x$, $m_z^{\bar{\omega}_z} < 0$ and $c_z^\beta < 0$. It is clear from (2.25) that especially with an increase of the mass distribution along the lengthwise axis of the fuselage, an increase in the aerodynamic damping of the pitch and the coefficient of lift, the value $\delta_{B_1}^z$ increases. At negative ω_p, m_z^β and c_z^β the coefficient $\delta_{B_0}^z$ is also positive. The greater the gyroscopic moment of the engine rotor the greater the value of this coefficient. /55

Usually the ratio $\Delta n_z/\Delta n_y < 0$, since the increase of Δn_y, as a rule, leads to a reduction of V and to a certain decrease in c_z, which [as we can see from (2.30)] in turn causes a decrease in n_z. If ω_{x_0} is then greater than zero, the term of the expression (2.47) which contains the partial derivative δ_B^z is negative. At $d\delta_B/dn_y < 0$, the term in (2.47) promotes an increase in the absolute value of this total derivative, i.e., an increase in the motion of the elevator per unit of vertical load factor.

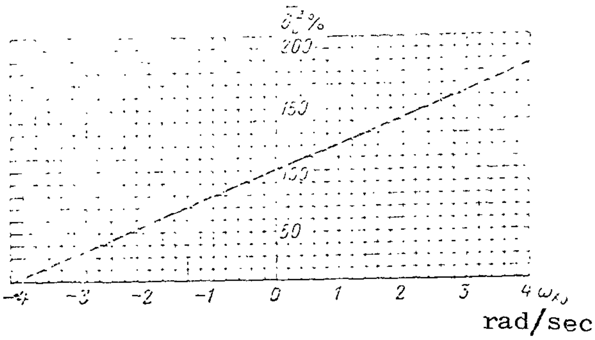


Fig. 2.3: Example of the Dependence of $\overline{\delta_B^z}$ on the Value of the Initial Angular Velocity of Roll.

The change in the relative value of the partial derivative $\delta_B^{n_z}$ can be illustrated by the example in Figure 2.3. Here the symbol $\overline{\delta_B^z}$ represents the relationship

$$\bar{\delta}_B^z = \frac{(\delta_B^z)\ \text{at}\ \omega_{x_0} \neq 0}{(\delta_B^z)\ \text{at}\ \omega_{x_0} = 0} \cdot 100\% = \frac{\delta_B^z}{\delta_{B_0}^z} 100\%.$$

(c) PARTIAL DERIVATIVE δ_B^V

The partial derivative δ_B^V, like δ_B^n, is the sum of three components:

$$\delta_B^V = \delta_{B_0}^V + \delta_{B_1}^V \omega_{x_0} + \delta_{B_2}^V \omega_{x_0}^2. \tag{2.56}$$

The coefficients that enter into it are expressed by the following formulas: /56

$$\delta^V_{B_0} = \frac{1}{V_0 m_z^{c_y}}\left(2c_{y_0}\varphi_n - M_0 m_z^M + \frac{M_0 c_z^M m_z^{c_y}}{c_z^{c_y}}\right); \quad (2.57)$$

$$\delta^V_{B_1} = \frac{1}{m_z^{c_y}}\left(\frac{b_A M_0 c_z^M m_z^{\bar{\omega}_z}}{V_0^2 c_z^{c_y}} + \frac{J_p \omega_p c_{y_0}\varphi}{J_z g c_y^{\alpha}}\right);$$

$$\delta^V_{B_2} = \frac{(J_x - J_y)\, c_{y_0}\varphi}{J_z g c_y^{\alpha} m_z^{c_y}} \quad (2.58)$$

$$\varphi = \varphi(M, c_y) - a_0 c_y^2 - M_0 c_y^M - 2c_{y_0}. \quad (2.59)$$

When $\phi < 0$ and $J_y > J_x$, the coefficient $\delta^V_{B_2}$ is negative. With the same assumption regarding the sign of ϕ and $\omega_p < 0$, the first term in (2.58) will also be negative. If

$$c_z^M < 0,\ c_z^{c_y} < 0 \text{ and } m_z^{\bar{\omega}_z} < 0,$$

the second term in this formula will be positive, but usually it turns out to be less than the modulus of the first term. In such cases, the coefficient $\delta^V_{B_1}$ will also be negative.

It is clear from (2.57) that with $m_z^M > 0$ and with the remaining negative aerodynamic derivatives, the coefficient $\delta^V_{B_0}$ which characterizes the longitudinal static stability by velocity in the absence of any rotation by the aircraft, is positive. Consequently, with $\omega_{x_0} > 0$, both terms in (2.56) which contain ω_{x_0} tend to make the partial derivative δ^V_B negative, while the term δ_{B_0} helps to keep the derivative δ^V_B positive, i.e., in this case the stability of the aircraft with velocity decreases with an increase in the angular velocity of roll ω_{x_0}.

A drop in the stability of the aircraft with velocity (a decrease in δ^V_B) naturally leads to a drop in the absolute value of the total derivative $d\delta_B/dn_y$ in the presence of stability of the aircraft under force, since the ratio $\Delta V/\Delta n_y < 0$ (with an increase in the vertical force, the speed of the aircraft is reduced).

Examples of the change in the relative value of the partial derivative δ^V_B and the ratio of the absolute values of the individual terms in (2.56) as a function of the angular velocity of roll ω_{x_0} are shown in Figures 2.4 and 2.5. The arbitrary symbols used in /57 these graphs represent the following values:

$$\bar{\delta}^V_B = \frac{(\delta^V_B) \text{ at } \omega_{x_0} \neq 0}{(\delta^V_B) \text{ at } \omega_{x_0} = 0} 100\% = \frac{\delta^V_B}{\delta^V_{B_0}} 100\%;$$

$$\bar{\delta}^V_{B_0} = \frac{\delta^V_{B_0}}{\sum |\delta^V_{B_i} \omega^i_{x_0}|} 100\%;$$

$$\bar{\delta}^V_{B_1} = \frac{\delta^V_{B_1} \omega_{x_0}}{\sum |\delta^V_{B_i} \omega^i_{x_0}|} 100\%;$$

$$\bar{\delta}^V_{B_2} = \frac{\delta^V_{B_2} \omega^2_{x_0}}{\sum |\delta^V_{B_i} \omega^i_{x_0}|} 100\%,$$

where

$$\sum_{i=0}^{2} |\delta^V_{B_i} \omega^i_{x_0}| = |\delta^V_{B_0}| + |\delta^V_{B_1} \omega_{x_0}| + |\delta^V_{B_2} \omega^2_{x_0}|.$$

After examining Figure 2.5, we conclude that in this example, the effect of the term with $\delta^V_{B_1}$ (the gyroscopic moment of the engine rotor, the derivative c^M_z, etc) on the value is negligibly small. Up to high values of the angular velocity of roll ω_{x_0}, the principal influence upon δ^V_B comes from the term $\delta^V_{B_0}$ (the effect of the static margin of longitudinal stability, the derivatives m^M_z, m^β_z, etc.). It is only at very high values of ω_{x_0} that the term of (2.56) which contains $\delta^V_{B_2}$ begins to play a significant role (effect of the mass distribution). Thus, in particular, even at ω_{x_0} = 1.5 rad/sec the relative value of this term amounts to only about 10%.

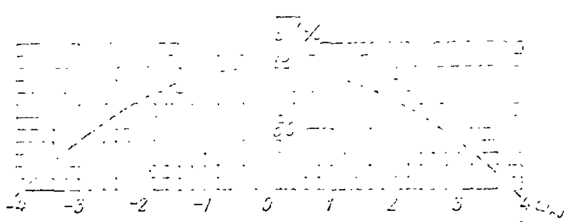


Fig. 2.4: Example of the Dependence of $\bar{\delta}^V_B$ on the Value of the Initial Angular Velocity of Roll.

To determine the required increase in the deviation of the control stick (Δx_B), we can write an expression which is analogous to (2.46): /

$$\Delta x_{B} = x_{B}^{n}\Delta n_y + x_{B}^{\dot n}\Delta \dot n_y + x_{B}^{\ddot n}\Delta \ddot n_y + x_{B}^{z}\Delta n_z + x_{B}^{\dot z}\Delta \dot n_z +$$

$$+ x_{B}^{\ddot z}\Delta \ddot n_z + x_{B}^{V}\Delta V + x_{B}^{\dot V}\Delta \dot V + x_{B}^{\ddot V}\Delta \ddot V.$$

The derivatives x_B^n, $x_B^{\dot n}$, ..., x_B^z, ..., $x_B^{\ddot V}$ involved here can be determined by using the familiar kinematic relationship:

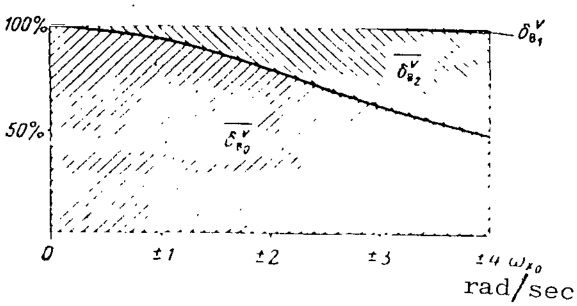


Fig. 2.5: Ratio of the Absolute Values of the Individual Terms in the Expression for δ_B^V.

$$\Delta x_B \approx dx_B = \int_{\delta_{B_0}}^{\delta_{B_1}} \frac{d\delta_B}{k_{t.r.}}$$

where $k_{t.r.}$ is the gear ratio of the system for longitudinal control of the aircraft.

The same method can be used to determine the increase of the forces acting on the control stick and its corresponding derivatives during manual longitudinal control.

in particular:

$$\frac{dP_B}{dn_y} = P_B^n.$$

The derivatives x_B^n and P_B^n are the basic characteristics of the longitudinal contraollability of the aircraft under force, which can be sensed directly by the pilot. The deviation of their actual values from the normal ones can create prerequisites for the unintentional entry of the aircraft into critical regimes.

2.3. Angular Velocities of Pitch Roll Coupling /59

The angular velocity of rotation of an aircraft, at which it loses its stability and undergoes a sharp deterioration of controllibility is called "critical". It is clear that the larger the absolute value of the critical angular velocity becomes, the smaller is the possibility of attaining it when performing a maneuver. Let us begin by considering the critical angular velocities of roll at which, as a result of an interaction between the longitudinal and lateral motions of the aircraft, it loses its longitudinal stability.

As we showed earlier, the absolute values of the partial derivatives δ_B^n and δ_B^V decrease with an increase in the absolute value of the angular velocity of roll ω_{x_0}, according to the law of the quadratic parabola. If $\omega_{x0} < 0$, the increase of its modulus (in the case studied above) also leads to a decrease in the partial derivative δ^z_B. The change of δ^z_B as a function of ω_{x_0} is linear. Hence, from the theoretical standpoint, there are always values of ω_{x_0} such that the total derivative $d\delta_B/dn_y$ becomes equal to zero. A further increase in the absolute values of ω_{x_0} causes this derivative to change sign and become positive, then to begin increasing. The aircraft becomes unstable under stress load factor with fixed controls.

The angular velocity of roll at which the total derivative $d\delta_B/dn_y$ becomes equal to zero is called the angular velocity of roll, pitch roll coupling ω^P_{cr}.

The value of ω^P_{cr} can be determined from (2.47) when the derivative $d\delta_B/dn_y$ is equal to zero. In this case, we obtain from (2.47):

$$\delta^n_B\,\Delta n_y + \delta^z_B\,\Delta n_z + \delta^V_B\,\Delta V = 0.$$

Substituting here the expressions for the partial derivatives δ^n_B, δ^z_B, and δ^V_B from (2.49), (2.53), and (2.56), respectively, and grouping the terms with the same degrees of angular velocity of roll, we obtain the total quadratic equation relative to ω^P_{cr}:

$$\begin{aligned} &A(\omega^P_{cr})^2 + B\omega^P_{cr} + C = 0,\\ &A = \delta^n_{B_2}\,\Delta n_y + \delta^V_{B_2}\Delta V;\\ &B = \delta^n_{B_1}\,\Delta n_y + \delta^z_{B_1}\,\Delta n_z + \delta^V_{B_1}\Delta V;\\ &C = \delta^n_{B_0}\,\Delta n_y + \delta^z_{B_0}\,\Delta n_z + \delta^V_{B_0}\Delta V. \end{aligned} \tag{2.60}$$

From this we obtain the angular velocity of roll which is critical for pitch: /60

$$\omega^P_{cr} = \frac{-B \pm \sqrt{B^2 - 4AC}}{2A}. \tag{2.61}$$

If the angular velocity of roll in flight in absolute value turns out to be larger than the critical value determined by (2.61), i.e., $|\omega_{x0}| > |\omega^P_{cr}|$, then (as we mentioned earlier) the aircraft loses its stability with load. When this occurs, a fundamental change occurs in the controllability characteristics of the aircraft, disorienting the pilot, considerably complicating his task, and frequently causing the aircraft to stall. This is explained by the following:

The pilot judges the maneuverability of the aircraft by its reaction to certain movements of the control surfaces, the degree and nature of the movements of the controls (stick and pedals), and changes in the force applied to the latter while in flight. By

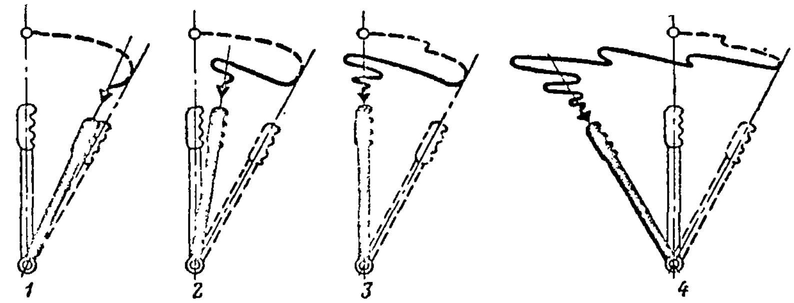


Fig. 2.6: Movements of the Control Stick, Required to Shift the Aircraft from One Stable Flight Regime to Another. (1) Stable Aircraft (Adequate Degree of Stability); (2) Stable Aircraft (Low Level of Stability); (3) Neutral Aircraft; (4) Unstable Aircraft: ---- = Forward Movement of Control Stick, O = Its Original Position, → = Backward Movement of Control Stick (Final Position Shown by Black Silhouette).

tilting the control surfaces, he overcomes the aerodynamic, inertial, and gyroscopic moments acting on the aircraft. Depending on the nature of the changes in these moments during disturbed motion of the aircraft, the movements of the controls (stick and pedals) required to carry out a certain maneuver will change with time.

In an unstable aircraft, when making a transition from one regime of stable flight to another, the pilot must first move the control stick (in the case of longitudinal instability) and the pedals (in case of lateral instability) in the direction of the maneuver being carried out; he must then move them back again and tilt them in the opposite direction to halt the rapidly developing motion of the aircraft (Fig. 2.6). In neutral and slightly stable /61 aircraft, this back-and-forth movement of the controls is continued when flying, but its magnitude is much smaller.

In a neutral aircraft, the pilot returns the controls to the initial position after completing a maneuver; in stable flight however, he keeps moving them back and forth. It is only when there is a sufficient static margin of stability in the aircraft that these dual motions (back and forth) of the controls are nearly absent, and the way in which they are moved for control becomes simpler and easier. The forces on the controls also change in a similar manner.

Not only the movements of the controls and the forces on them, as applied by the pilot in performing a certain maneuver, but also the nature of the actual transition process and the lag in the reaction of the aircraft to the motion of a given control (the stick) are different in stable, neutral and unstable aircraft, i.e., the ease of controlling the aircraft depends to a large extent on its stability.

Stability improves the reaction of the aircraft to the actions of the pilot, reduces the lag, and facilitates control over the flight regime and the "dosing" of the required movements of the controls to maintain the required flight regime and to carry out required maneuvers. This considerably relieves the load on the pilot's attention, reduces the demands on his mental and physical energies, and cuts down on the amount of flight time devoted merely to maintaining a required flight regime. The picture is quite the opposite in an unstable aircraft. Hence, it is much more difficult to pilot an unstable aircraft than a stable one.

For the frequent case of flight with $\Delta n_y \neq 0$ at $\Delta n_z = \Delta V = 0$, (2.61) becomes

$$\omega^P_{cr} = \frac{-\delta^n_{B_1} \pm \sqrt{(\delta^n_{B_1})^2 - 4\delta^n_{B_2}\delta^n_{B_0}}}{2\delta^n_{B2}} \tag{2.62}$$

If we substitute the values δ^n_{B0}, δ^n_{B1}, and $\delta^n_{B_2}$ from (2.50), (2.51), and (2.52), respectively, we will have:

$$\omega^P_{cr} = -\frac{J_p\omega_p}{2(J_x - J_y)} \pm \sqrt{\left[\frac{J_p\omega_p}{2(J_x - J_y)}\right]^2 + \frac{Kc_y^\alpha\sigma_n}{(J_x - J_y)\mu_{inst}}} \tag{2.63}$$

With $\sigma_n < 0$, both terms under the radical sign in (2.63) become positive ($J_y > J_x$). Therefore, the right-hand term in (2.63) is always greater in absolute value than the left-hand term. Consequently, in this particular case one of the solutions will always be $\omega^P_{cr} > 0$, and the other will be $\omega^P_{cr} < 0$. Of these two values of ω^P_{cr} determined by (2.62), the positive ω^P_{cr} has the smaller absolute value at $\omega_p < 0$. In this case (flight at $\omega_{x_0} > 0$), the critical angular velocity of roll represents the difference between the first and second terms. In flight at $\omega_{x_0} < 0$, the critical angular velocity of roll is the algebraic sum of the two terms (in this case, there is a minus sign in front of the radical. The difference in the absolute values of the negative and positive ω^P_{cr} in this case is due solely to the effect of the gyroscopic moment.

From (2.63) in particular, we can see that the increase in the mass distribution along the longitudinal axis of the fuselage, as well as the decrease in c_y^α, lead to a decrease in the absolute values of ω^P_{cr}, i.e., they constitute the prerequisites for an earlier loss of stability by the aircraft under stress. At $\omega_p < 0$, in the case where $\omega_{x_0} < 0$, the increase in the absolute value of the gyroscopic moment leads to an increase in the absolute value of

ω^P_{cr}; in the case where $\omega_{x_0} > 0$, however, the picture is the opposite. In the general case of an intentional maneuver in space with $\Delta n_y \neq 0$, $\Delta n_z \neq 0$, and $\Delta V \neq 0$, the angular velocities of roll, critical for pitch, are usually close in absolute value to the values of ω^P_{cr} determined by (2.62) and (2.63). Therefore, for the sake of approximation we can assume the values of ω^P_{cr} determined by (2.63) to be angular velocities of roll critical for pitch also in the general case of an intentional maneuver in space, i.e., from the condition of equating the partial derivative δ^n_B to zero. If it is necessary to have a more precise determination of the value of ω^P_{cr} for an intentional maneuver in space, this must be done with the aid of (2.61), substituting in it the values of Δn_y, Δn_z, and ΔV, determined from the motion equations of the center of gravity of the aircraft in the maneuver in question.[2]

2.4. Movement of the Rudder and Stresses on the Pedals per Unit of Sideslip Angle.

(a) MOVEMENT OF THE RUDDER PER UNIT OF SIDESLIP ANGLE

The increase in the movement of the rudder, or movable stabilizer, required for a given change in flight regime, is found from (2.21):

$$\Delta\delta_H = \frac{1}{M_y^{\delta_H}}\Big[J_y\dot{\omega}_y + (J_x - J_z)\,\omega_{x_0}\omega_z - M_y^{c_y}\Delta c_y - M_y^{\beta}\beta - M_y^{\omega_y}\omega_y - M_y^{\omega_x}\omega_x - M_y^{\dot{\beta}}\dot{\beta} - M_y^{V}\Delta V + J_p\omega_p\omega_z\Big] \qquad (2.64)$$

By substituting into this equation the changes in the angular velocities of pitch, yaw and roll from (2.43), (2.44) and (2.45), respectively, then replacing Δc_y and β by means of (2.25) and (2.38) and finally differentiating the expression obtained for β, we obtain the following equation for determining the movement of the rudder per unit of sideslip angle: /63

$$\delta_H^{\beta} = \frac{1}{M_y^{\delta_H}}\left[-M_y^{\beta} + M_y^{\omega_y}\frac{Z^{\beta}}{mV_0} + J_p\omega_p\omega_{x_0} + (J_x - J_z)\,\omega_{x_0}^2\right], \qquad (2.65)$$

i.e.,

$$\delta_H^{\beta} = \delta_{H_0}^{\beta} + \delta_{H_1}^{\beta}\omega_{x_0} + \delta_{H_2}^{\beta}\omega_{x_0}^2 \qquad (2.66)$$

The values of δ^{β}_{Hi} (i = 0, 1, 2) used in (2.66) are determined by

[2]See for example: Kotik, M.G., A.V. Pavlov, et al: Letnyye ispytaniya samoletov (Flight Testing of Aircraft) "Mashinostroyeniye", 1965.

the formula:

$$\delta^{\beta}_{H_0} = -\frac{\sigma_6}{m_y^{\delta_H}}, \qquad (2.67)$$

where

$\sigma_6 = m_y^{\beta} - \frac{c_z^{\beta} m_y^{\bar{\omega}y}}{2\mu_6}$ — is the static margin of lateral stability of the aircraft under lateral stress, and

$\mu_6 = \frac{2m}{\varrho_H S l}$ — is the relative rate of the aircraft in lateral movement;

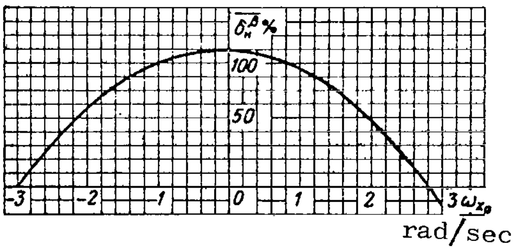


Fig. 2.7: Example of the Dependence of $\overline{\delta^{\beta}_{H}}$ on the Value of the Initial Angular Velocity of Roll.

$$\delta^{\beta}_{H_1} = \frac{J_p \omega_p \mu_6}{K m_y^{\delta_H}} \qquad (2.68$$

and

$$\delta^{\beta}_{H_2} = \frac{(J_x - J_z)\mu_6}{K m_y^{\delta_H}} \qquad (2.69)$$

As we can see from (2.69), with $Jz > Jx$ and $m_y^{\delta H} < 0$, the value of δ^{β}_{H2} becomes greater than 0. If $\omega_p < 0$, the value of δ^{β}_{H1} is also positive. From (2.67) is follows that with negative aerodynamic derivatives, the coefficient δ^{β}_{H0} becomes less than zero. Figure 2.7 shows the degree of change in the relative value of the partial derivative δ^{β}_{H} as a function of the value of the angular velocity of roll ω_{x0}. An example of the change in the relative relationship of the absolute values of individual terms on the right-hand side of (2.66) as a function of ω_{x0} is given in Figure 2.8. Here the following conditional symbols are employed: /61

$$\overline{\delta^{\beta}_{H}} = \frac{(\delta^{\beta}_{H}) \text{ for } \omega_{x_0} \neq 0}{(\delta^{\beta}_{H}) \text{ for } \omega_{x_0} = 0} 100\% = \frac{\delta^{\beta}_{H}}{\delta^{\beta}_{H_0}} 100\%;$$

$$\overline{\delta^{\beta}_{H_0}} = \frac{|\delta^{\beta}_{H_0}|}{\sum |\delta^{\beta}_{H_i} \omega^{i}_{x_0}|} 100\%;$$

$$\overline{\delta^{\beta}_{H_1}} = \frac{|\delta^{\beta}_{H_1} \omega_{x_0}|}{\sum |\delta^{\beta}_{H_i} \omega^{i}_{x_0}|} 100\%;$$

$$\overline{\delta^{\beta}_{H_2}} = \frac{|\delta^{\beta}_{H_2} \omega^{2}_{x_0}|}{\sum |\delta^{\beta}_{H_i} \omega^{i}_{x_0}|} 100\%,$$

where

$$\sum_{i=0}^{2} |\delta^{\beta}_{H_i}\omega^{i}_{x_0}| = |\delta^{\beta}_{H_0}| + |\delta^{\beta}_{H_1}\omega_{x_0}| + |\delta^{\beta}_{H_2}\omega^{2}_{x_0}|.$$

It is clear from Figure 2.8 that in the example given, the influence of the term with δ^{β}_{H1} (the influence of the gyroscopic moment of the engine rotor) on the value is small. At small and average values of the angular velocity of roll, the principal effect on the derivative δ^{β}_{H} is exerted by the term in (2.66) which contains the coefficient δ^{β}_{H0} (the effect of the static margin of lateral stability of the aircraft under lateral force. It is only at very large angular velocities of /65 roll that the effect of the term with δ^{β}_{H2} (the effect of the mass distribution over the aircraft) becomes significant.

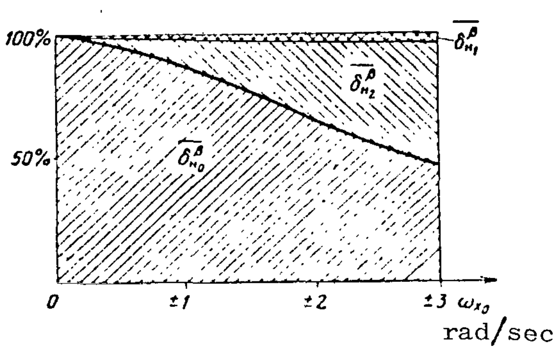


Fig. 2.8: Ratio of the Absolute Values of Individual Terms in the Expression for δ^{β}_{H}.

Clearly, the travel (movement) of the pedals per unit of sideslip angle in accordance with (2.66) can also be represented in the form:

$$x^{\beta}_{H} = x^{\beta}_{H_0} + x^{\beta}_{H_1}\omega_{x_0} + x^{\beta}_{H_2}\omega^{2}_{x_0}.$$

(b) CHANGE IN FORCE ON THE PEDALS PER UNIT OF SIDESLIP ANGLE

The increase in the force on the pedals ΔP_{H}, in controlling flight without the aid of boosters, is equal to:

$$\Delta P_{H} = A_{H}\left[\Delta\delta_{H} + \frac{m^{\beta}_{\mathrm{sm}}}{m^{\delta_H}_{\mathrm{sm}}}\left(\beta + \frac{\bar{L}_{\mathrm{v.s}}l}{V_0\sqrt{k_{\mathrm{v.s.}}}}\omega_y\right)\right], \qquad (2.70)$$

where

$$A_{H} = -k_{\mathrm{sm}} S_{H} b_{H} k_{\mathrm{v.s}} p_{\mathrm{us}} \frac{n_{y_0}}{c_{y_0}} m^{\delta_H}_{\mathrm{sm}};$$

$k_{\mathrm{v.s.}} = \frac{V^2_{\mathrm{v.s.}}}{V^2_0}$ — is the coefficient of drag in the vicinity of the vertical section of the tail;

$p_{\mathrm{us}} = \frac{G}{S}$ — is the specific load on the wing;

$V_{v.s.}$ — is the rate of oncoming airflow in the vicinity of the vertical section of the tail;

$m^{\beta}_{s.m.}$ — is the derivative of the rudder hinge moment coefficient according to the sideslip angle;

$m^{\delta_H}_{s.m.}$ — is the derivative of the rudder hinge moment coefficient according to the angle of its deviation;

S_H and b_H are the area and chord of the rudder, respectively.

$\bar{L}_{v.s.}=\frac{L_{v.s.}}{l}$ — is the relative arm of the vertical section of the tail.

If we substitute the value of ω_y from (2.44) into (2.70) and differentiate the expression thus obtained for β, we will have the equation for determining the change in force on the pedals per unit of sideslip angle P^{β}_{H}:

$$P^{\beta}_{H}=P^{\beta}_{H_0}+P^{\beta}_{H_1}\omega_{x_0}+P^{\beta}_{H_2}\omega^2_{x_0}, \tag{2.71}$$

where

$$P^{\beta}_{H_0}=A_H\left[\delta^{\beta}_{H_0}+\frac{m^{\beta}_{s.m.}}{m^{\delta_H}_{s.m.}}\left(1-\frac{\bar{L}_{v.s.}c^{\beta}_z}{\mu_6\sqrt{k_{v.s.}}}\right)\right]; \tag{2.72}$$

$$P^{\beta}_{H_1}=A_H\delta^{\beta}_{H_1}; \tag{2.73}$$

$$P^{\beta}_{H_2}=A_H\delta^{\beta}_{H_2}. \tag{2.74}$$

With $m^{\delta_H}_{s.m.} < 0$, the value of A_H is positive. Consequently, in this case the values of $P^{\beta}_{H_2}$ and $P^{\beta}_{H_1}$ will have the same sign as the coefficients $\delta^{\beta}_{H_2}$ and $\delta^{\beta}_{H_1}$. As mentioned above, the value of $\delta^{\beta}_{H_0}$ is usually negative. If c_z is then less than zero, and $m^{\beta}_{s.m.} > 0$, then both terms within the brackets in (2.72) will have the same sign, so that the absolute value of $P^{\beta}_{H_0}$ will increase. The reverse is the case when $m^{\beta}_{s.m.} < 0$: the swiveling moment of the rudder causes a reduction in the degree of static directional stability of the aircraft with free controls.

The values x^{β}_{H} and P^{β}_{H} are some of the most important characteristics of the maneuverability of the aircraft along a path, involving the sideslip term. The latter plays an essential role in the possibility of unintentional entry of the aircraft into critical regimes (more detail on this will be given later).

2.5. Angular Velocities of Roll, Yaw-Roll Coupling

An analysis of (2.66) and (2.71) will show that in this case the increase in the absolute value of the angular velocity of roll leads to a decrease in the absolute values of the derivatives δ^{β}_{H}

and P_H^β. As a rule, there are always values of ω_{x_0} (both negative and positive) such that these derivatives turn out to be equal to zero. A further increase in the absolute values of ω_{x_0} leads to a change of sign of the derivatives δ_H^β and P^β from negative to positive, i.e., the aircraft becomes directionallv unstable (according to the restricted or free control over its path).

By analogy with the above, we shall call the angular velocity of roll at which the aircraft loses its directional stability in flight with restricted control (the derivative δ_H^β becomes equal to zero), the "critical angular velocity for yaw with restricted flight direction", and will represent it by $\omega^p_{cr.r}$. The value ω_{x_0} at which the aircraft loses its directional stability in flight with free control ($P_H^\beta = 0$), will be referred to as the critical angular velocity of roll for yaw with free path control, and will be represented by $\omega^p_{cr.f}$. The value $\omega^p_{cr,r.}$ is found from (2.66), by making the derivative δ_H^β equal to zero. Solving the quadratic equation thus obtained for the angular velocity of roll, we will have:

$$\omega^p_{cr.r.} = \frac{-\delta^\beta_{H_1} \pm \sqrt{(\delta^\beta_{H_1})^2 - 4\delta^\beta_{H_2}\delta^\beta_{H_0}}}{2\delta^\beta_{H_2}} \qquad (2.75)$$

From this, with the aid of (2.67), (2.68), and (2.69), we obtain: /67

$$\omega^p_{cr.r.} = \frac{J_p\omega_p}{2(J_z - J_x)} \pm \sqrt{\frac{J_p^2\omega_p^2}{4(J_x - J_z)^2} + \frac{K\sigma_6}{(J_x - J_z)\mu_6}}. \qquad (2.76)$$

At negative aerodynamic derivatives ($\sigma_b < 0$) and $J_z > J_x$, both terms under the radical sign are positive. Consequently, the first term in (2.76) is smaller than the second in absolute value. Therefore the solution of (2.76) must necessarily be one positive and one negative value of $\omega^p_{cr.r.}$. But since the first term in (2.76) is negative when $\omega_p < 0$, then the positive value of $\omega^p_{cr.r.}$ will be smaller in absolute value than the critical angular velocity of roll in this particular case, and the negative $\omega^p_{cr.r.}$ will be larger in absolute value. It is particularly clear from (2.76) that the greater the mass distribution along the longitudinal axis of the fuselage, the smaller will be the degree of static directional stability of the aircraft with restricted control and aerodynamic yaw (the smaller the absolute value of σ_b), the smaller the absolute values of $\omega^p_{cr.r}$). An increase in the absolute value of the gyroscopic moment of the engine leads to an increase in the absolute value of the negative $\omega^p_{cr.r.}$ (flight with $\omega_{x_0} < 0$) and a decrease in the positive $\omega^p_{cr.r.}$ (flight with $\omega_{x_0} > 0$).

Now let us find the formula for determining the critical angular velocity of yawing with free directional control. From (2.71) with $P_{H}^{\beta} = 0$, we will have

$$\omega_{\text{crit}}^{p} = \frac{-P_{H_1}^{\beta} \pm \sqrt{(P_{H_1}^{\beta})^2 - 4P_{H_2}^{\beta} P_{H_0}^{\beta}}}{2P_{H_2}^{\beta}}. \tag{2.77}$$

The angular velocities of roll critical for pitch and yaw are sometimes called the first and second critical velocities. Usually, the one which is smaller is called the first critical velocity and the larger one is called the second critical velocity. Theoretical analysis and flight tests show that the danger of an undesirable interaction between the longitudinal and lateral motions of the aircraft when performing a maneuver with a roll ($\omega_{x_0} \neq 0$) increases as the difference between the absolute values of the first and second critical velocities increases (as the absolute value of the first critical velocity decreases and the absolute value of the second increases).

CHAPTER 3

STALL

/68

3.1. Distinctive Features of Supersonic Aircraft which Determine the Characteristics of Critical Regimes

The aerodynamics, design and weight of modern supersonic aircraft differ considerably from those of subsonic aircraft.

The principal reasons for the considerable difference between the characteristics of supersonic and subsonic aircraft are the following:

(1) The change to a sweptback or delta wing configuration and tail.

(2) A decrease in the length and relative thickness of these structural elements.

(3) An increase in the wing loading.

(4) An increase in the length of the fuselage and the nose section in particular.

(5) An increase in the volume of the fuselage and the weight load placed upon it.

(6) The use of a controlled stabilizer and (in some cases) a controlled tail fin.

(7) The use of high-powered jet engines.

(8) The incorporation of boosters and automatic devices in the control system.

(9) Equipping the aircraft with special devices.

Characteristic trends in the development of individual aircraft parameters can be traced by using the example of the well-known Soviet aircraft (typical jet fighters) built by the Special Design Office headed by A.I. Mikoyan: the first mass-produced Soviet jet aircraft, the MIG-9; the MIG-15 subsonic jet aircraft; the sonic MIG-17; the first Soviet supersonic aircraft, the MIG-19; the first

Soviet aircraft with a controlled stabilizer, the MIG-19s; and the MIG-21, which can travel at high supersonic speeds. Table 3.1 lists the following characteristics of these aircraft: the sweepback angle of the wing, χ_w; the sweepback angles of the horizontal and vertical sections of the empennage, $\chi_{h.e.}$ and $\chi_{v.e.}$, respectively; /69
the mean aerodynamic chord of the wing, b_A; the maximum flying weight of the aircraft, G_{max}; the ratio of the axial moments of inertia of the aircraft, the length of the fuselage λ_f, the relative length of the nose section of the fuselage $\bar{l}_{n.f.} = l_{n.f.}/L_f$ (where L_f is the length of the fuselage and $l_{n.f.}$ is the length of its nose section); the relative area of the elevator $\bar{S}_e$; and the relative length of the horizontal section of the empennage $\bar{L}_{h.e.}$

TABLE 3.1.

RELATIVE CHARACTERISTICS OF SUBSONIC AND SUPERSONIC AIRCRAFT

Characteristic \ Aircraft	MIG-9	MIG-15	MIG-17	MIG-19	MIG-19s	MIG-21
χ_w in deg	0	35	45	55	55	57
$\chi_{h.e.}$ in deg	20	40	45	55	55	55
$\chi_{v.e.}$ in deg	30	56,0	55.68	57.5	57.5	60.5
b_A in m	1.52	2.12	2.36	3.02	3.02	4.0
G_{max} in kg	4990	4810	5200	6820	7560	7840
$J_x:J_z$	1:1.75	1:2.90	1:3.71	1:4.51	1:4.33	1:11.095
$J_x:J_y$	1:2.04	1:3.23	1:4.23	1:5.30	1:5.12	1:11.80
λ_f	6.26	5.57	6.35	7.19	7.19	9.82
$\bar{l}_{n.f.}$	0.360	0.440	0.465	0.495	0.495	0.527
$\bar{S}_e$	0.228	0.264	0.285	0.197	0.572	0.589
$\bar{L}_{h.e.}$	2.88	2.5	2.46	1.42	1.64	1.32

A clear idea of the characteristic trends in the change of the external appearance of aircraft with a change to supersonic speeds can be obtained from Figures 3.1 and 3.2. Figure 3.3 shows the change in the ratio of the planar moments of inertia I_z/I_x, the wing loading $P_l = G/S$, (for the mean flying weight of the aircraft), and the wing length λ for a number of aircraft, as a function of their year of manufacture.

Let us examine in somewhat greater detail, the basic features of supersonic aircraft of conventional design (fuselage-wing-empennage) which affect the characteristics of the critical regimes. The modern supersonic aircraft of conventional design is characterized by wings with small relative thickness and a high sweepback angle (along the leading edge in the case of a delta wing and along the quarter-chord line in the case of a sweptback wing). The wings are much shorter in supersonic aircraft then in subsonic ones (see Fig. 3.3). This means a reduction of the value $c_{y_{max}}$ and the /70
absolute values $c_{y_{min}}$, as well as a considerable increase in the

absolute values of the critical angles of attack in supersonic aircraft (Fig. 3.4). The shape of the curve $c_y = f(\alpha)$ in the range of near-critical angles of attack in supersonic aircraft is smoother than for subsonic aircraft; this is due primarily to the shorter wings (with respect to the wings of subsonic aircraft).

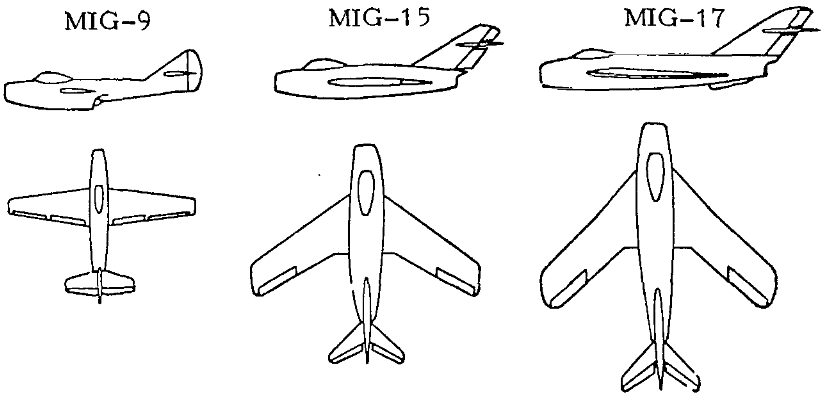


Fig. 3.1.: Outlines of Subsonic Aircraft (MIG-9 and MIG-15) and the Supersonic MIG-17.

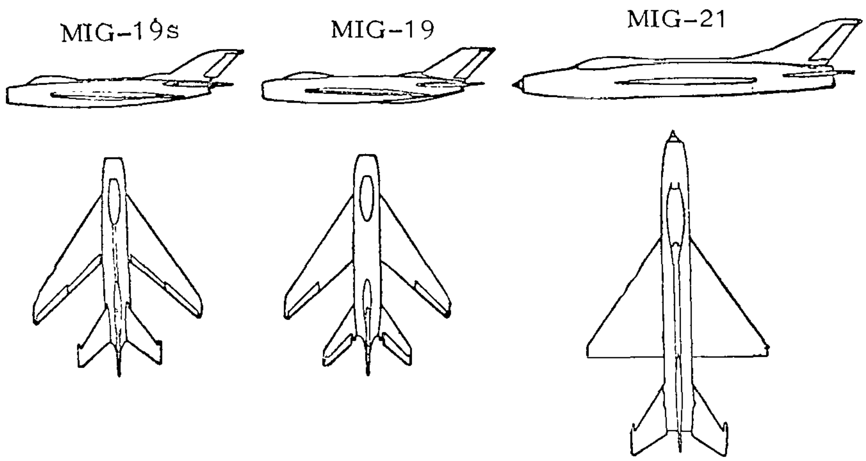
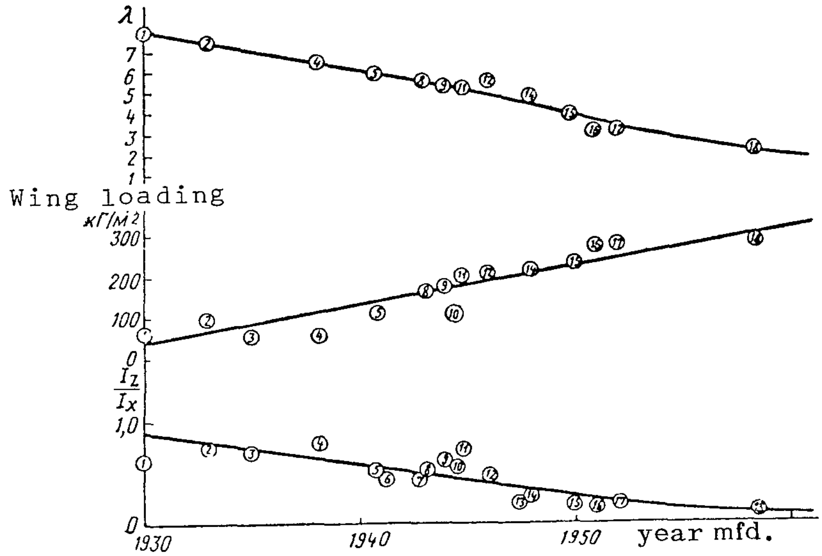


Fig. 3.2.: Outlines of Supersonic Aircraft (MIG-19, MIG-19s, and MIG-21).

The large sweepback angle of the wing has a considerable effect on the rolling characteristics of the aircraft. The empennage of supersonic aircraft, like the wings, is thin and has a large sweep- /71
back angle. The dimensions of the elements for longitudinal control (and in aircraft with a controlled tail fin, lateral control as well) are much larger in these planes than are the corresponding elements in subsonic aircraft (see Table 3.1.).

Changes of this type in the external appearance also entail modifications in the inertial characteristics of the aircraft. A reduction in the internal volume of the wings means that practically



1: R-5 2: I-14 3: I-16 4: UT-2 5: Yak-1 6: MIG-3
7: LA-5 8: Yak-9 9: R-39 10: LAGG-3 11: LA-11
12: MIG-9 13: LA-15 14: MIG-15 15: MIG-17 16: MIG-19
17: MIG-19s 18: MIG-21

Fig. 3.3: Ratio of Planar Moments of Inertia, Wing Loading, and Wing Length in Various Aircraft of Soviet Manufacture.

all of the fuel, equipment, and payload of a supersonic aircraft are located in the fuselage. Increase in length with simultaneous reduction of the relative thickness of the fuselage, accompanied by a shift of considerable weight to the latter, results (as we have said) in a sharp increase in the weight distributed along the longitudinal axis ox_1 in supersonic aircraft (see Fig. 3.3., where the increase in weight along the longitudinal axis of the aircraft is characterized by a drop in the ratio I_z/I_x). This means that the weight distributed along the wingspan is decreased to a considerable degree. This weight distribution also entails a significant difference in the inertial characteristics of supersonic aircraft with respect to the same characteristics in subsonic aircraft. The ellipsoid of inertia in modern supersonic aircraft is much more elon- /72
gated with respect to the longitudinal axis. Due to the high weight concentration in the longitudinal axis, a supersonic aircraft in a spin (and occasionally, in a stall) can become subject to very high inertial moments of yaw and especially pitch, while the inertial moment of roll is relatively small.

The flying weight and wing loading of a supersonic aircraft are much greater (see Table 3.1 and Fig. 3.3).

Due to the requirements of supersonic aerodynamics, the tapering (diffusive power) of the tail section of the fuselage of a supersonic aircraft is reduced or completely absent, i.e., the tail section of the fuselage is thicker relative to the tail section

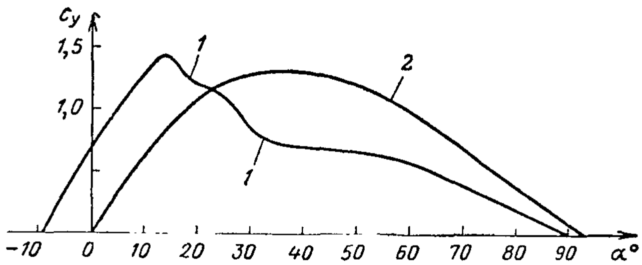


Fig. 3.4.: Examples of Curves $c_y = f(\alpha)$ in a Subsonic Aircraft (1) and a Supersonic Aircraft (2).

of the fuselage in subsonic aircraft. The relatively high degree of tapering of the wing (especially in the case of a delta-winged aircraft) and its low relative thickness produce an increase in the length of the chords at the base of the wing. All of the above, as well as an increase in the sweepback angle of the empennage, increase the aerodynamic shadowing of the latter by the wing and fuselage at high angles of attack.

The conditions of aeronautical operation of supersonic aircraft also show extensive differences relative to subsonic aircraft. There has been considerable expansion of the range of altitudes, speeds, and Mach numbers. The operational altitudes and speeds (Mach numbers) at which modern aircraft fly has generally increased. The thrust of supersonic aircraft is very great. In many instances, the maximum thrust of the engine even exceeds the weight of the plane. The maximum speeds for rectilinear horizontal flight of supersonic aircraft usually exceed the permissible limit over a large portion of the operational altitude range. All of these factors mean that at practically all operational flight speeds at these altitudes (in a range of Mach numbers from the lowest permissible value $M_{min\ per}$ to the threshold value M_{thresh}), there is a relatively wide safety margin for longitudinal stresses $n_x \gg 0$ (Fig. 3.5) and therefore for longitudinal accelerations as well ($\dot{V} \gg 0$, since $\dot{V} = g(n_x - \sin\theta)$. However, the flight duration of modern supersonic aircraft at operational altitudes is relatively short. In this connection, supersonic aircraft are also generally involved in unsteady, rapidly changing flight patterns. /73

All of the factors enumerated above have a significant influence on the characteristics of maneuverability, stability (see Chapters I and II) and details of the critical regimes of supersonic aircraft. Let us examine these details.

3.2. Details of Supersonic Aircraft Stalling

(a) TENDENCY TO STALL

Regardless of the existence of a number of factors which reduce the tendency of modern supersonic aircraft to stall (which will be discussed below), these planes nevertheless have just as great a possibility of unintentionally falling into critical regimes as did the old subsonic aircraft. Such a peculiar aspect of the behavior of supersonic aircraft can be explained by three main reasons:

(1) The increase in the operational ranges of altitude, speed, and Mach number, as well as the increase in the wing loading. Thus, for example, as the flight altitude and wing loading increases, the available reserve for c_y decreases, i.e., there is an increase in the likelihood of the aircraft unintentionally falling below $c_{y\,st}$ in performing a maneuver. At high supersonic Mach numbers, due to the decrease in the degree of directional stability of the aircraft (see ChapterI), very high sideslip angles may result, corresponding to a drop in $c_{y\,st}$, along with the appearance of a violent interaction between the longitudinal and lateral motions of the aircraft, leading first to a loss of stability and then stalling of the aircraft.

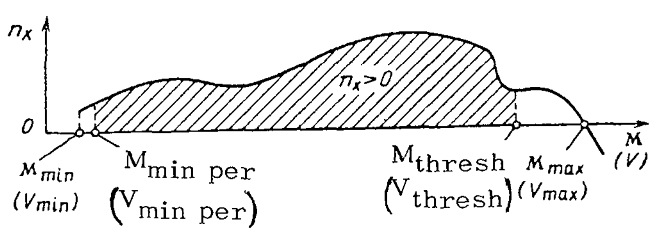


Fig. 3.5.: Example of Change in Longitudinal Force as a Function of Mach Number for a Supersonic Aircraft Flying Horizontally in a Straight Line.

(2) A considerable complication of the pilot's tasks in view of the number of automatic or semiautomatic guidance and navigational systems. In this case, the pilot must practically simultaneously keep track of a great many instruments and devices which indicate flight parameters and operating conditions of the engine and other systems and assemblies of the aircraft; at the same time, these flight regimes are unstable by nature and can change very rapidly. Supersonic aircraft usually have a great many peculiar features of stability, maneuverability and flying characteristics, as well as a large number of diverse limitations (with respect to flying regimes, operational regimes of the engine, etc.) that is greater than in subsonic aircraft. These factors make additional demands on the pilot's attention. /74

(3) Certain characteristic features of stability and maneuverability (existence of instability during load factor at high angles of attack, etc.; see Chapter I), as well as specific behavior of the aircraft under special flight conditions (for example, if the booster systems for control break down).

Under battle conditions, when the pilot cannot watch his instruments carefully and often performs complicated maneuvers, and also when flying under difficult meteorological conditions, the likelihood of unintentional aircraft stalling is increased.

The factors enumerated above increase the likelihood that a supersonic aircraft will stall. On the other hand, there are many factors which reduce the tendency of supersonic aircraft (compared to subsonic aircraft) to stall. The following are the most important ones:

Stalling of supersonic aircraft usually occurs much more gradually under the same conditions than in subsonic aircraft. With a lack of initial sideslip and essentially equal conditions, supersonic aircraft stall less "willingly" than subsonic ones. This is explained primarily by the smooth outline of the curve of the relationship $c_y = f(\alpha)$ in regions of near-critical angles of attack and a reduction in the values of $c_{y\max}$, as well as by somewhat greater values for the moments of inertia relative to the longitudinal axis J_x in supersonic aircraft. In addition, the motion of the control stick required to attain $c_{y\text{st}}$ in supersonic aircraft is usually much greater than in subsonic aircraft. Therefore, unintentional stalling at low (close to or equal to $V_{\min}$) speeds in rectilinear horizontal flight without sideslip, at relatively low altitudes under normal operational conditions, is encountered much less often in supersonic aircraft than in subsonic ones. These then are the factors which decrease the tendency of supersonic aircraft to stall.

In view of the fact that the stalling of aircraft at negative angles of attack occurs relatively rarely (since inverted flight or as it is sometimes called, "flying upside down", is a rare occurrence in normal operational flight of aircraft), we shall discuss only the cases of stalling in the initial regime of normal flight, i.e., flight in which the pilot is in a "head up" position (in contrast to the "head down" position while "flying upside down"). Later on, when discussing spin, we will show how and under what conditions a supersonic aircraft can attain negative near-critical and critical angles of attack. /75

The following are stalling (or breakdown) characteristics of aircraft:

(1) Symptoms signaling the onset of a stalling regime.

(2) Minimum flight speeds with different weights and external configurations of the aircraft, operational regimes of the engine,

flight altitudes, etc., the values of $c_{y_{st}}$ for various external configurations of the aircraft, Mach numbers of flight, operational regimes of the engine, etc.

(3) Behavioral characteristics of the aircraft at large subcritical angles of attack, and during a stall.

(4) Techniques of piloting required to avert stalling as well as to pull the aircraft out of a stall under the usual operational conditions of flight (the conditions for establishing an operational flight regime).

Let us consider the characteristics of stalling in supersonic aircraft. We shall begin with the behavioral characteristics of the aircraft during a stall (details of its behavior at high subcritical angles of attack were discussed earlier, in Chapter I).

(b) VARIETIES OF STALL

As we know, stalling of an aircraft occurs as the result of the appearance of areas of disrupted flow on the wing at large angles of attack. It is usually accompanied by dropping of the nose and/or rolling of the aircraft. Symmetric occurrence and development of areas of disrupted flow on the right and left wings with increase of the angle of attack, due to reduction of the lifting force of the wing, causes the aircraft to go into a nosedive. In this case, α_{st} is very close to, or equal to, α_{cr} ($\alpha_{st} \approx \alpha_{cr}$). In the case of highly unsymmetric development of areas of disrupted flow on the right and left wings, there is an initial tendency for the aircraft to go into a sideslip, which is then followed by the dropping of the nose of the plane. A stall of this kind occurs at α_{st} much less than α_{cr} ($\alpha_{st}<\alpha_{cr}$). The dropping of the nose of the aircraft in this instance is due to an upset of the balance of forces acting in the vertical direction (the lift becomes less than the weight of the aircraft), as well as by the sideslip toward the lower wing which occurs during rolling (the aerodynamic moment of yaw attempts to overcome the sideslip and forces the nose of the aircraft downward).

When the angular velocity of roll is nonzero ($\omega_x \neq 0$), the angle of attack along one wing will increase and that on the other will decrease. Such an increase in the angle of attack along one wing during rotation will cause the local areas of disrupted flow to increase more intensely. In other words, the increase in the coefficient of lift of the aircraft with the increase of the angle of attack (beginning at near-critical angles of attack) can occur much more slowly and be accompanied by a drop in the value of α_{cr} (see Fig. 3.6.). The asymmetric location of the areas of disrupted flow along the wing, produced by the development of the angular velocity of roll, leads in turn to a further increase in the absolute value of this angular velocity of roll, etc. /76

The characteristics of stalling are not only a function of the point of origin of the flow disruption (for example, at the tip or base of a wing), but also of the speed and nature of the development of areas of disrupted flow on the upper wing surface. If the aircraft itself and the flow conditions were ideally symmetric, the distribution of the areas of disrupted flow would be the same on both wings and would not cause the aircraft to roll in a stall. However, under actual conditions, there is always some degree of non-symmetry in the aircraft and the flow around it, so that there is consequent asymmetry in the disruption of the flow, which leads to the development of rolling moments. If the areas of disrupted flow on both wings develop symmetrically until the aircraft reaches angles of attack close to α_{st}, and asymmetry develops only when the transition to α_{st} takes place, stalling will occur with dropping of the nose and practically simultaneous rolling of the aircraft (simultaneous rolling and pitching moments of high amplitude). In this case, $\alpha_{st} \leq \alpha_{cr}$.

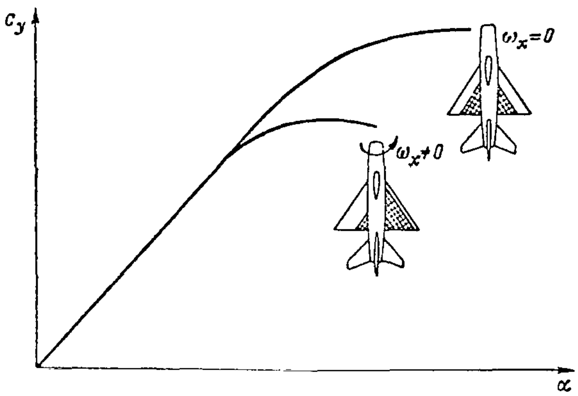


Fig. 3.6.: Change in Lifting Capacity of an Aircraft During Rotation (Areas of Disrupted Flow on the Upper Surface of the Wing are Shaded).

The latter form of stall also usually occurs in supersonic aircraft. It is caused by the non-symmetric origin and development of areas of disturbed flow on a wing with a large sweepback angle and short length, so that at high speeds the distribution of the disruption over the wing usually causes the development of relatively large rolling (as well as twisting) aerodynamic moments As a result, the plane enters a spiral trajectory, or a tailspin. /77

The occurrence of relatively high rolling and twisting moments at high rates of distribution of areas of disrupted flow over the wing is explained as follows. Let us say that an area of disrupted flow appears earlier on the right wing than on the left. Due to the high rate of propagation, this area of disrupted flow expands rapidly and soon may occupy a large part of the right wing, while on the left the area of disrupted flow has not yet arisen during this brief time interval. As a result, the lift of the right wing decreases considerably, while its drag increases significantly. Hence, despite the relatively small size of the wing, supersonic aircraft generate relatively greater aerodynamic moments, which are caused by such nonsymmetric distribution of areas of disturbed flow. There is a rolling moment (on the right wing, in the case described above) and a twisting moment (in the direction of the right wing).

However, the angles of roll which are developed during stalling of aircraft with sweptback wings are relatively smaller in absolute value than for aircraft with wings that are not swept back. The latter phenomenon is related to the appearance of large moments of recovery from rolling, caused by an increase in the degree of lateral static stability of aircraft with sweptback wings large angles of attack and inhibition of the development of a rolling motion.

(c) EFFECT OF SIDESLIP

In supersonic aircraft with sweptback or delta wings, even a slight degree of sideslip can have a significant effect on the shape of the curve $c_y = f(\alpha)$. In subsonic aircraft with wings at right angles to the fuselage, this effect is much less pronounced (Fig. 3.7.). This explains the fact, well-known from practical flying experience and flight testing, that aircraft with swept-back wings can stall at angles of attack smaller than α_{st} in aircraft with the wings at right angles to the fuselage, regardless of the fact that the critical angles of attack (in the presence of skidding) are usually much greater in the case of the former type of aircraft.

The much greater effect of sideslip on the aerodynamic characteristics of the sweptback wing is due primarily to the fact that the nature of the flow around it (distribution of pressure over the wing) is much more subject to external influences than is the case /78
for wings at right angles to the fuselage. The increased sensitivity of the sweptback wing to external influences is related to an important characteristic of its flow: the shift of the boundary layer spanwise toward the wingtips. This leads to a swelling of the boundary layer at the wingtips and consequently to the appearance of areas of slightly compressible or even incompressible flow. Hence, even relatively weak external influences can lead to significant changes in the flow picture.

In addition, the occurrence of sideslip is accompanied by changes

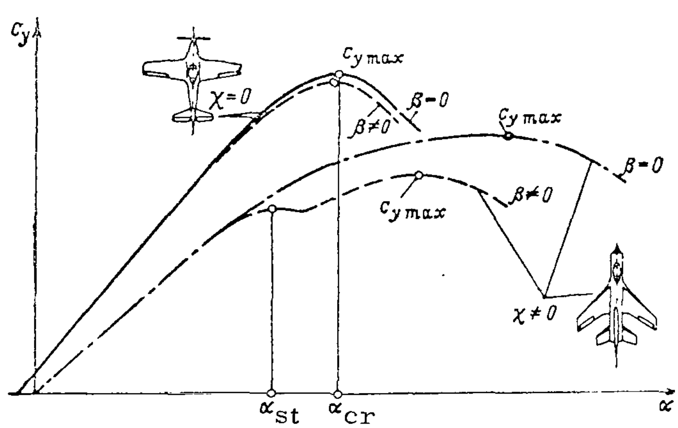


Fig. 3.7.: Effect of Sideslip on the Shape of the Curve $c_y = f(\alpha)$.

in the effective length and effective sweepback of the right and left sweptback wings. This produces extensive changes in the aerodynamic characteristics of both wings (see Chapter IV).

(d) THREE FORMS OF THE CURVE $c_y = f(\alpha)$.

The behavior of an aircraft in a stall is determined mainly by the nature of the relationship $c_y = f(\alpha)$ at angles of attack $\alpha \geq \alpha_{st}$. The more sharply the slope of this curve changes with the angle of attack (the partial derivative $a_y = \partial c_y/\partial\alpha$), the more abruptly the aircraft goes into a stall. The curves $c_y = f(\alpha)$ of various aircraft can be divided into three types on the basis of the abruptness of the change of the derivative a_y in the region of near-critical angles of attack. Here are their characteristics, listed separately (Fig. 3.8.):

(1) The presence of a clearly pronounced maximum in the curve $c_y = f(\alpha)$ with an intense variation (not abrupt, but rather smooth) /79
in the coefficient of lift with a transition to critical angles of attack.

(2) The presence of a sharp maximum, with a very steep (practically vertical) drop in the coefficient of lift when the critical angle of attack is exceeded. In this case, the derivative a_y remains practically constant over the entire range of subcritical angles of attack.

(3) The absence of a clearly marked maximum in the curve $c_y = f(\alpha)$, when the values of the coefficient of lift are relatively close to $c_{y\max}$ over a wide range of near-critical angles of attack.

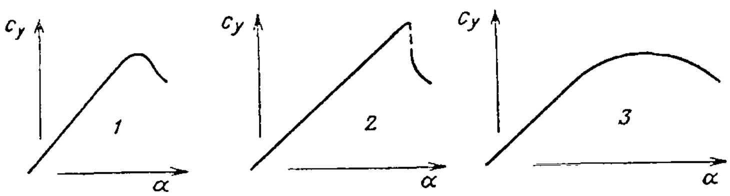


Fig. 3.8. Three Types of the Curve $c_y = f(\alpha)$.

The curve of $c_y = f(\alpha)$ of the first type is characteristic of subsonic aircraft, while the third type is typical of supersonic aircraft. The second type is encountered in old subsonic aircraft, with relatively thick wings at right angles to the fuselage.

The much smoother appearance of the curve $c_y = f(\alpha)$ in the vicinity of the maximum and the smaller values of $c_{y\max}$ for supersonic aircraft are due mainly to the relatively short wings as well as to their relative thinness.

Shorter wings are more prone to the occurrence of early flow turbulence; small local areas of turbulent flow occur on them much earlier (at angles of attack which are much smaller than in the case of long wings). In addition, the unloading peaks on the upper surface of a wing of this type are smaller. Therefore, when developed areas of disrupted flow appear and the unloading peaks mentioned above (which are related to them) disappear, the pressure differential (change in the resultant aerodynamic force of the wing) is no longer as great, and takes place more evenly, then in long wings.

The ratio of the perimeter to the area of the wing, in the case when the latter is short, is greater than when it is long. Hence, in short wings the effect of flow at the edges on the nature of flow over the entire wing, increases. This produces a drop in the negative pressure peaks and an additional (but important) gradual drop in negative pressure on the upper surface of the wing as the angle of attack increases. Thanks to the small peaks of negative /80
pressure, the earlier and more intense turbulence of the flow, and the smaller pulsations, the flow over shorter wings is more stable when the angle of attack increases. Therefore, a complete (or nearly so) disruption of flow on a short wing (with appearance of developed areas of disrupted flow, occupying practically the entire wing, or in any case a large part of it), is involved and arises at much greater angles of attack.

(e) ABRUPTNESS OF STALL

The more abruptly stalling occurs, the more dangerous it is, since the pilot has less time to combat the stall and keep the plane from going into a spin. It is also perilous because the aircraft loses its customary attitude during the stall and assumes attitudes which are not permissible in normal flying operations; this is especially true for aircraft that are not highly maneuverable, and becomes more of a problem as the abruptness of the stall increases. The abruptness of the stall, i.e., the intensity of the occurrence of the involuntary motion of the aircraft after the angle of attack α_{st} has been exceeded in the initial portion of this motion is characterized by the maximum attainable absolute value of the angular acceleration of rolling $\varepsilon_{x\max} = \dot{\omega}_{x\max}$ and the maximum absolute value of the angular velocity of the roll which then arises, $\omega_{x\max}$. These are the two most important characteristics of stalling. Besides these, stalling is also characterized by the absolute values of the maximum angular velocity of yaw $\omega_{y\max}$ and the maximum angular acceleration of yaw $\varepsilon_{y\max} = \dot{\omega}_{y\max}$. These four parameters determine the operating conditions to a considerable degree, as well as the possibility of orienting the pilot in a dangerous regime that evolves rapidly.

The average absolute values $\omega_{x\max}$ and $\varepsilon_{x\max}$ of old and modern aircraft, obtained on the basis of data from their flight tests at altitudes on the order of 5 to 8 km, are compared in Table 3.2.

hese values were obtained with the control surfaces set for a stall elevator deflected completely upward, rudder turned fully in the irection of the turn), and neutral ailerons. The materials in the able confirm the fact that the stalling of supersonic aircraft nder comparable initial conditions actually occurs more smoothly han in subsonic aircraft. We will show later on that the values $\varepsilon_{x\max}$ and especially $\omega_{x\max}$ can vary within wide limits in one and he same aircraft, depending on the initial flight conditions.

In Figure 3.9, the strip charts from flight recorders aboard a ubsonic MIG-9 and a supersonic MIG-21, showing data obtained during light in a stall with further transition to spin, are reproduced or comparison. In both cases, the aircraft sideslip to the right /81 nd the nose yawed to the right as well. Approximately 3.5 seconds fter the beginning of the stall, the MIG-9 (see Fig. 3.9,a) reached he maximum absolute angular velocities of roll and yaw $\omega_{x\max} \approx 2.65$ rad/sec and $|\omega_{y\max}| \approx 0.7$ rad/sec. In the case of stalling of the IIG-21, (see Fig. 3.9,b) values of $\omega_{x\max} \approx 0.65$ rad/sec and $|\omega_{y\max}| \approx 0.45$ rad/sec were reached approximately 4 sec later.

TABLE 3.2

MAXIMUM ANGULAR VELOCITIES AND ACCELERATIONS IN STALLING OF VARIOUS AIRCRAFT

Aircraft \ Characteristics of stall	$\lvert\omega_{x\max}\rvert$ rad/sec	$\lvert\varepsilon_{x\max}\rvert$ rad/sec^2
UT-1	0.85	0.65
Yak-7	1.8	1.2
Yak-9B	1.4	1.1
P-39("aircobra")	2.0	2.2
LAGG-3*	1.5	1.3
LAGG-3**	0.9	0.85
ULA-7	0.7	1.45
ULA-7***	1.1	1.5
LA-9	1.5	0.6
LA-11	1.9	1.3
MIG-9	2.6	1.05
MIG-15	1.0	0.65
MIG-17	0.9	0.9
MIG-19	0.8	0.75
MIG-21	0.5	0.38

*without flaps **with flaps ***with flaps closed

As we have already mentioned, stalling in supersonic aircraft usually causes them to describe a spiral. This means that during the first seconds after the stall begins, there arises a moment of pitch which tends to raise the nose of the aircraft somewhat (in the example shown in Fig. 3.9,a the angular velocity of the pitch was positive for approximately 5 seconds after the stall began: $t \approx 5$ to 9 sec), after which time the diving moment made its appearance (in Fig. 3.9,a $|\omega_{z\,max}| \approx 0.55$ rad/ sec at t = 8 sec).

For an approximate estimate of the ratio of the values of the angular acceleration in a stall of this kind, let us examine the qualitative picture of the change in the aerodynamic forces and moments acting on an aircraft in this regime. /82

When relatively extensive areas of disrupted flow appear on the wing, the vector of the resultant aerodynamic force of the aircraft increases somewhat at first, then decreases and shifts toward the rear along a chord. The initial increase in the resultant aerodynamic force is caused by the increase of its coefficients with an increase in the angle of attack, while further decrease is due to the interruption of the increase of aerodynamic coefficients and a drop in the flight speed as a result of the rise in the value of the coefficients of vertical (c_{y_1}) and tangential (c_{x_1}) aerodynamic forces with the angle of attack in the case of an aircraft with sweptback wings is given in Figure 3.10.

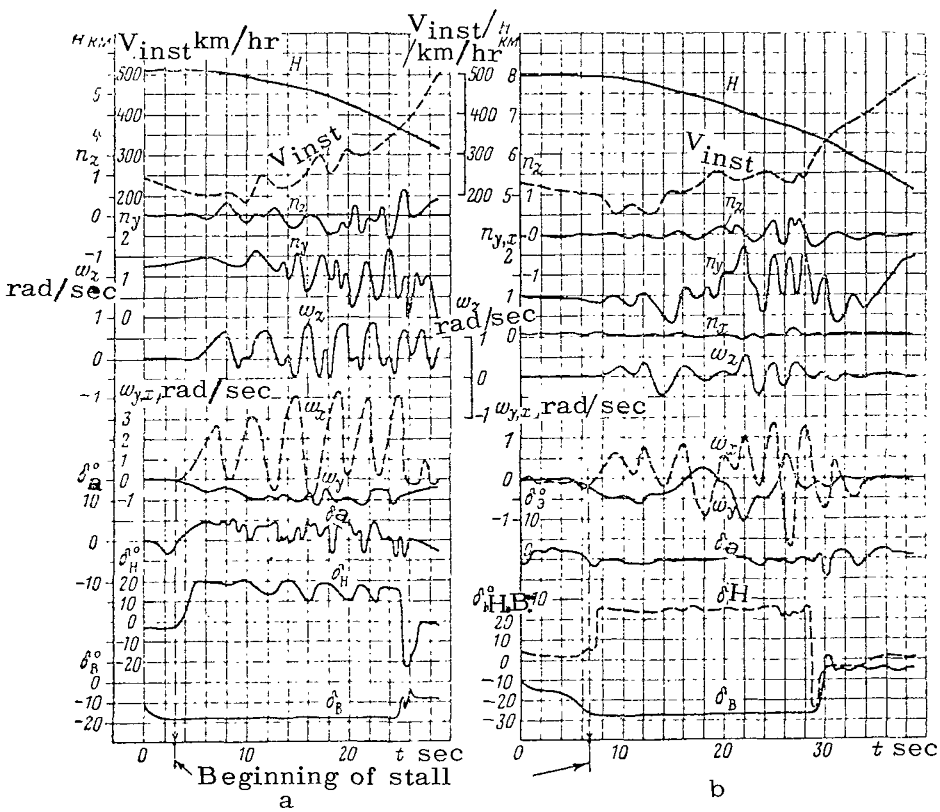


Fig. 3.9: Stall with Transition to Spin in the MIG-9 (a) and MIG-21 (b).

Let us represent the vector of the resultant aerodynamic force before the stall, i.e., before the appearance of developed areas of disrupted flow) by $\vec{R}$ (its components in a system of axes with the coordinates X and Y, related to the aircraft). Let us also use $\vec{R}_{st}$ to represent the same value after the stall, i.e., after disruption of the flow has occurred (its components are X_{st} and Y_{st}). We can represent the difference between these vectors as follows: $\Delta\vec{R}_{st}$ $(\Delta X_{st};\ \Delta Y_{st}) = \vec{R}(X;\ Y) - \vec{R}_{st}$ $(X_{st};\ Y_{st})$. The vector $\vec{R}_{st}$ usually forms a small angle with the vertical to the plane of the wing chords, so that its components along the axes of the system of coordinates related to the aircraft satisfies the inequality /83

$$X_{st} \ll Y_{st}$$

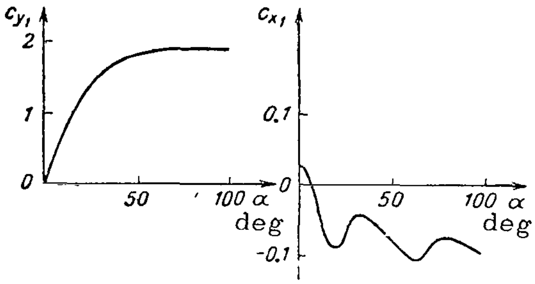


Fig. 3.10.: Change in Coefficients of Vertical and Tangential Aerodynamic Forces- (c_{y1} and c_{x1}) with the Angle of Attack in Aircraft with Sweptback Wings.

The tangential and vertical forces (n_x and n_y, respectively) acting on the aircraft during a stall with the engine cut out ($P = 0$) will be equal to

$$n_x = \frac{X_{st}}{G} < 0,$$

$$n_y = \frac{Y_{st}}{G} > 0,$$

so that $|n_x| \ll n_y$.

With a symmetric disruption of flow, the aerodynamic resultant shifts along the axis oz_1 to the value z_1. This causes the appearance of the aerodynamic moments of roll $M_{x st} = Y_{st} z_1$ and yaw $M_{y st} = X_{st} z_1$, constituting the angular accelerations of roll and yaw

$$\varepsilon_x = \dot{\omega}_x = \frac{M_{x st}}{J_x} = \frac{Y_{st} z_1}{J_x}$$

and

$$\varepsilon_y = \dot{\omega}_y = \frac{M_{y st}}{J_y} = \frac{X_{st} z_1}{J_y}$$

Since $J_y \gg J_x$ in supersonic aircraft, and $Y_{st} \gg X_{st}$, then $|\varepsilon_x| \gg |\varepsilon_y|$ in a stall. In this case, the motion of the aircraft is an intense rolling in the direction of the wing with the more developed (or, more exactly, greater in area or displaced further toward the wingtip) area of disrupted flow and a relatively less intense yawing of the nose of the aircraft in the direction of the same wing. The increase in the moments of inertia relative to /84 the vertical axis J_y in supersonic aircraft has led to a significant reduction in their angular acceleration, and consequently in the angular velocities of yaw or stall as well (as compared to subsonic aircraft).

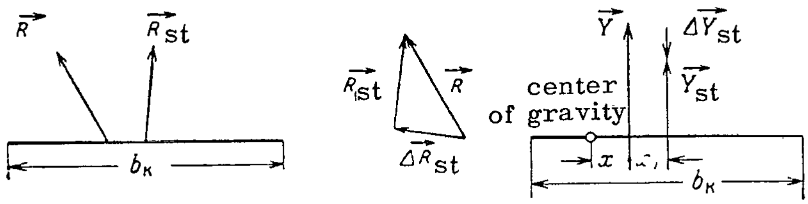


Fig. 3.11.: Diagram of Forces Acting on an Aircraft During a Stall.

We can use the diagram in Figure 3.11 to estimate the degree of angular acceleration of pitch. Prior to the stall, the aircraft is subject to the aerodynamic moment of pitch, which is equal to the product of the vertical component of the resultant aerodynamic force $\vec{Y}$ times the distance from the point where it is applied to the center of gravity of the aircraft $\vec{Y}x$. After stalling has begun, the aircraft is subject to the resultant aerodynamic moment of pitch, equal to the product of the force $\vec{Y}_{st}$ times its arm: $\vec{Y}_{st}$ $(x + x_1)$. The increase in the arm of the acting force in the second case is explained by the shift in the vector of the resultant aerodynamic force toward the rear along the chord after the aircraft has begun to stall. Hence, when the stall begins, the aerodynamic moment of pitch which was acting earlier on the aircraft changes in value:

$$\Delta M_{z_{st}} = Yx - Y_{st}(x + x_1) = \Delta Y_{st}x - Y_{st}x_1.$$

The increase of the moment of pitch to $\Delta M_{z_{st}}$ will change the angular acceleration of the pitch to the value

$$\varepsilon_z = \dot{\omega}_z = \frac{\Delta M_{z_{st}}}{J_z} = \frac{\Delta Y_{st}x - Y_{st}x_1}{J_z}$$

At $\Delta Y_{st} < 0$, which is usually the case when stalling has developed (due primarily to the loss of flight speed) and even at $\Delta Y_{st} > 0$, if only $|Y_{st}x_1| > |\Delta Y_{st}x|$, the aircraft has a tendency to drop its nose in the stall. We mentioned earlier that at the beginning of stall we usually have $\Delta Y_{st} > 0$. If we then have $|Y_{st}x_1| < |\Delta Y_{st}x|$, /8
the aircraft then raises its nose at the start of the stall, and it is only later (when the sign of ΔY_{st} changes) that a tendency to dive develops. Usually $J_z >> Jx$, while $|x_1| < |z_1|$, so that as a rule $|\varepsilon_z| < |\varepsilon x|$ during a stall.

In aircraft with highly swept wings, which do not have special anti-turbulence fittings, the disruption of flow which arises at the wingtips, being located far behind the center of gravity of the aircraft, can lead to a shift of the resultant aerodynamic force forward, to an extent which may be so great that it is even shifted ahead of the center of gravity. In this case, it is possible to have the production of high values of ε_z (angular velocities of pitch) due to the development of instability in the aircraft during stress at large angles of attack (see Chapter I).

In the example given above for the stalling of a supersonic aircraft (see Fig. 3.9,b) the average values for the angular accelerations were as follows:

$$\varepsilon_x \approx 0.38 \text{ rad/sec}^2,$$
$$\varepsilon_y \approx -0.12 \text{ rad/sec}^2,$$

i.e., we actually have $|\varepsilon_x| \gg |\varepsilon_y|$. The mean angular acceleration of pitch in this case was originally directed toward raising the nose of the aircraft and amounted to $\varepsilon_z = 0.12$ rad/sec^2, but then the value ε_z changed sign and became $\varepsilon_z \approx -0.24$ rad/sec^2. Hence, in this example we also have $|\varepsilon_z| < |\varepsilon_x|$.

(f) EFFECT OF WING SHAPE AND POSITION OF THE AXIS OF ROTATION OF THE AIRCRAFT

The effect of the wing shape at the point where the areas of disrupted flow develop can be illustrated clearly by the experimental data of Prof. B.T. Goroshchenko, which are listed in Figures 3.12 and 3.13. These graphs are schematic representations of the areas of disrupted flow on trapezoidal and sweptback wings. The graphs show the changes in the coefficients of lift of wing sections $c_{y\,\mathrm{sec}}$ in amplitude (dashed lines on the graphs) for two values of the angle of attack of the aircraft: α_1 and α_2 ($\alpha_2 > \alpha_1$). Also shown are the values of the coefficients of lift of wing sections $c_{y\,\mathrm{sec}}$ in which flow disruption arises in the section under consideration. It is clear that flow disruption occurs on the wing in that section in which the value of $c_{y\,\mathrm{sec}}$ first becomes equal to $c_{y\,\mathrm{st}}$. We can see from the graphs that the increased tapering of the wing and the effect of giving the wing a sweptback shape permits shifting the areas where flow disruption occurs toward the wingtips.

With the appearance of developed areas of flow separation on the wing at subcritical angles of attack, the curve of $c_y = f(\alpha)$ for an aircraft with sweptback wings can sometimes have the shape shown in Figure 3.14. In this case, beginning with the angle of attack of the aircraft $\alpha = \alpha_{\mathrm{drop}}$, the slope of the curve $c_y = f(\alpha)$ decreases significantly the so-called "drop" appears in the curve), i.e., the influence of the flow separation considerably reduces the

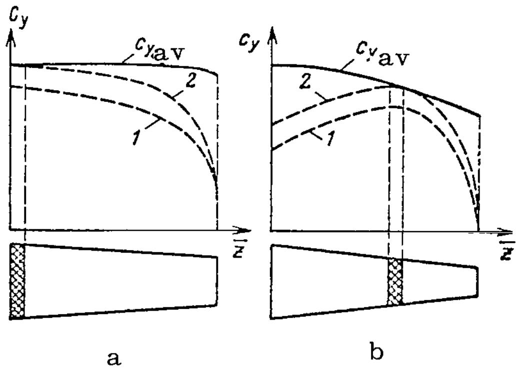


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Fig. 3.12.: Schematic Representation of the Location of Areas of Development of Flow Disruption on a Trapezoidal Wing with Slight (a) and Considerable (b) Taper

$1 - c_{y\,sec} = f(\bar{z})$ at $\alpha = \alpha_1$;
$2 - c_{y\,sec} = f(\bar{z})$ at $\alpha = \alpha_2\ (\alpha_2 > \alpha_1)$

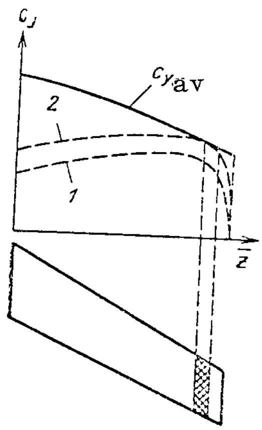

Fig. 3.13: Diagram of the Location of Area Where Flow Disruption Occurs on a Sweptback Wing with Slight Taper

$1 - c_{y\,sec} = f(\bar{z})$ at $\alpha = \alpha_1$;
$2 - c_{y\,sec} = f(\bar{z})$ at $\alpha = \alpha_2\ (\alpha_2 > \alpha_1)$

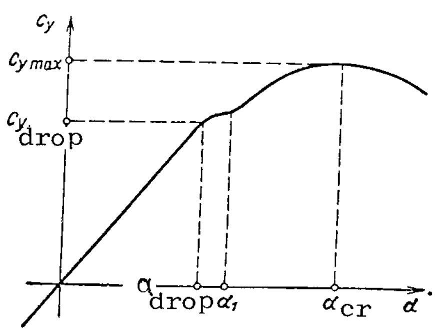

Fig. 3.14: Example of Curve $c_y = f(\alpha)$ with a "Drop" in the Curve at $c_y = c_{y\,drop}$

carrying power of the wing. However, beginning with an angle of attack $\alpha = \alpha_1$, as a result of local stabilization of areas of stall the carrying power of the wing again improves somewhat and the slope of the curve $c_y = f(\alpha)$ increases. In such cases, the value α_{drop}, determined from the point where the curve begins to drop, is the limit of the operational range of positive angles of attack of the aircraft. /87

However, this usually means that the angle of attack of the stall α_{st} is much less than α_{cr}, so that in flight with zero roll velocity and no sideslipping, it is practically impossible for the aircraft to attain $c_{y\max}$ (except when carrying out maneuvers at a high initial value of $\varepsilon_z > 0$). This is explained by the fact that the nose of the aircraft rises simultaneously with the start of the stall (when the aircraft attains $c_{y\,st}$). Even with the control stick pulled all the way back, the angle of attack of the aircraft still does not increase until the angular roll velocity of the leading edge develops during the stall, creating an increase of the local angles of attack on the descending wing (for details, see Chapter IV). Rolling of the aircraft also leads to sideslipping. It is only as the result of the development of an angular roll velocity $\omega_x \neq 0$ and a sideslip angle $\beta \neq 0$ (in the first case in question) that it is possible for the aircraft to attain first the critical and then the supercritical angles of attack.

In connection with the significant increase in the mass distribution along the longitudinal axis of the fuselage and the corresponding increase in the moments of inertia of the aircraft J_y, the rotation of supersonic aircraft in a stall occurs relative to an axis close to axis ox_1, related to the aircraft (see Chapter I). In stalls involving subsonic aircraft, the axis of rotation is located closer to the axis of the vector of the flight speed. This leads to an abrupt change in the angles of attack and sideslip during a stall, and hence to the appearance of a much less uniform (more oscillatory) motion of supersonic aircraft in this regime.

(g) WARNING BUFFETING AND SHAKING OF THE AIRCRAFT

The warning (or as it is sometimes called, the aerodynamic) buffet which is felt when the aircraft goes into a stall is of extreme importance to the pilot.

The warning buffet is very evident to the pilot, who feels it as a shock to the structure of the aircraft or as a shaking of individual parts of it, caused by the appearance of pulsating aerodynamic forces when stall occurs. In subsonic aircraft, transition to $c_{y\,st}$ usually is accompanied by the appearance of warning buffets, since there is a considerable difference between the values of $c_{y\,j}$ and $c_{y\,st}$. The values of $c_{y\,j}$ and $c_{y\,st}$ in supersonic aircraft in a stall from minimum (or nearly so) flight speeds (low Mach numbers, /88
$n_y \approx 1$), are usually very close to or even equal to each other. The intensity of the warning jolting in supersonic aircraft in case

of stalling in such flight regimes is usually slight. At low initial flight speeds (Mach numbers), sometimes even the transition to $c_{y\,st}$ fails to cause enough of a buffet to the structure so that e the pilot feels it, i.e., the warning buffeting is completely absent.

When supersonic aircraft with sweptback or delta wings enter transitional regimes, there is usually a rocking of the aircraft from one wing to the other (transverse oscillation), followed by rolling to one side. Sometimes these transverse oscillations are accompanied by oscillation of the aircraft along its flight path, associated with a deterioration of the characteristics of lateral stability with an increase in the angle of attack of the aircraft, as well as by the influence of pulsating aerodynamic forces, caused by the appearance of local areas of stall.

With entrance into a stalling regime with low initial Mach numbers and load factors ($n_{y_0} \approx 1$), these oscillations arise at values of the coefficient of lift of the aircraft which are less than $c_{y\,st}$ (usually by 15 to 20%), and take the form of slight periodic oscillations of the aircraft from one wing to the other with maximum angular roll velocities of the leading edge on the order of $\omega_x = \pm 3$ to 5 degrees/sec. This rocking of the aircraft persists until the stall begins. Usually the value of $c_{y\,per}$ in such aircraft is chosen to correspond to the beginning of these oscillations (in the absence of any other kind of limitations on the operational range of the values of the coefficient of lift).

3.3. Effect of Initial Conditions on the Characteristics of Stalling

All stalling regimes can be divided into three groups on the basis of the initial conditions: stalling at a minimal (or nearly so) speed of rectilinear horizontal flight, stalling at $V \gg V_{min}$ in a coordinated ($\beta_0 = 0$) vertical maneuver, and stalling in an arbitrary maneuver in space (in the general case at $\beta_0 \neq 0$ and $\Omega_0 \neq 0$).

(a) LOSS OF SPEED

Stalling from an initial regime of coordinated rectilinear horizontal flight when $V_{inst} \approx V_{min}$ ($n_{y0} \approx 1$) can occur as the result of smooth braking of the aircraft with a relatively slow backward movement of the control stick, i.e., with a gradual and slow braking to a minimum velocity (until the aircraft reaches $C_{y\,st}$). Such a motion of the aircraft is called speed loss (Fig. 3.15), the strip charts from flight recorders). In this case, the braking of the aircraft was accomplished with a neutral (initially balanced) position of the rudder and ailerons. /8

In a stall of this type, supersonic aircraft do not usually go into a spin spontaneously. Movement of the control stick forward from a neutral (or nearly so) position immediately after such a stall, as a rule, sends the aircraft into a dive, i.e., with complete restoration of normal maneuverability of the aircraft. Some supersonic aircraft do not enter a stall regime even when the control stick is moved all the way forward in the original flight regime

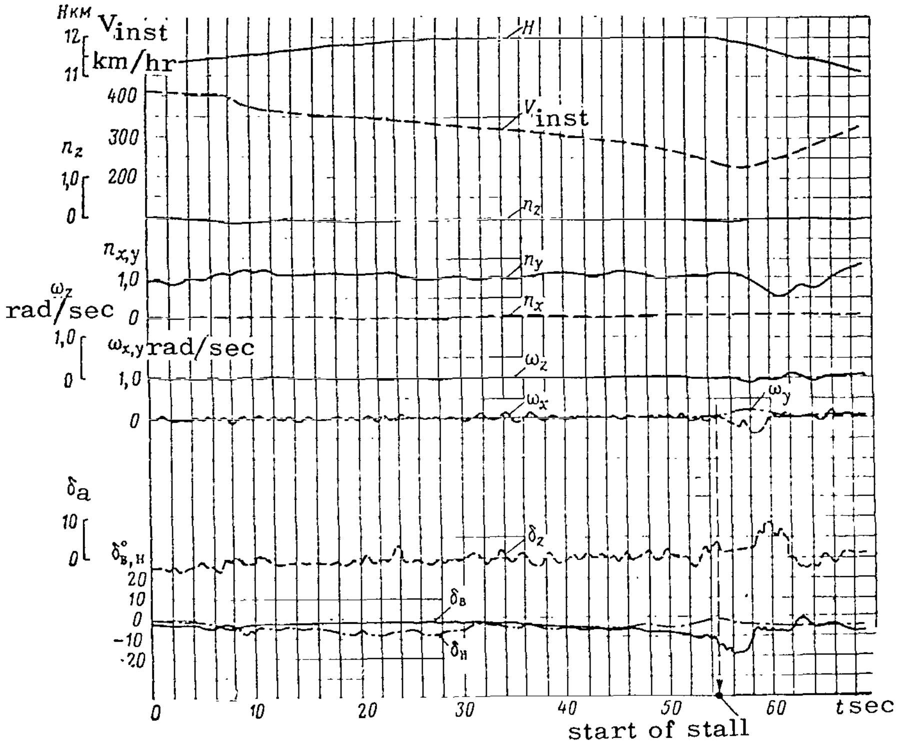

Fig. 3.15: Stalling of Aircraft as a Result of Loss of Flight Speed.

under consideration. During the initial sideslip ($\beta_0 \neq 0$) the stall with $V_{inst} \approx V_{min}$ occurs much more rarely and as a rule leads to a subsequent entrance of the aircraft into a spin.

(b) STALLING DURING COORDINATED MANEUVERS. /90

A stall in a coordinated vertical maneuver ($n_{y0} > 1$; $\beta_0 = 0$), i.e., a stall at relatively high instrument flight speeds ($V_{inst} \gg V_{min}$) without sideslip occurs very much less often than in the case of loss of speed (data on flight tests are presented in Fig. 3.16). This is explained by the high aerodynamic forces and moments acting on the aircraft at high instrumental flight speeds (reference pressures). A stall of this kind occurs when the control stick is manipulated in a coordinated vertical maneuver with the rudder and ailerons in the neutral (initially balanced) position.

Depending on the initial conditions, the values $|\omega_{x_{max}}|$ and

$|\varepsilon_{x\max}|$, as well as $|\omega_{y\max}|$ and $|\varepsilon_{y\max}|$ in the case of stalling of a given aircraft can vary within very wide limits. In the presence of an initial load factor $n_{y0} \gg 1$, i.e., in the case of a stall at high instrument speeds, these parameters increase significantly (Table 3.3), due mainly to the increase in the aerodynamic moments of roll and yaw caused by the increase in reference pressure.

TABLE 3.3

EFFECT OF INITIAL LOAD FACTOR ON STALLING CHARACTERISTICS (H_0 6 TO 7km).

Stalling characteristics / Aircraft	$\lvert\omega_{x_{\max}}\rvert$ rad/sec		$\lvert\varepsilon_{x_{\max}}\rvert$ rad/sec^2	
	$n_{y_0}\approx 1$	$n_{y_0}\approx 2{,}5-3{,}0$	$n_{y_0}\approx 1$	$n_{y_0}\approx 2{,}5-3{,}0$
MIG-19	0 8	2 0	0 75	2 7
MIG-21	0 65	1 7	0 38	2 0

The greater the initial Mach number when stalling occurs (i.e., the greater the flying altitude at a given instrumental rate of stall), the sooner the warning buffet usually occurs and the greater is its intensity. In supersonic aircraft, it often happens that at very low (practically zero) initial Mach numbers, the warning buffet is absent; however, as the initial Mach number increases, the buffet appears and becomes really quite intense in the case of a stall at sonic speeds.

In general, as far as supersonic aircraft versus subsonic aircraft are concerned (with the same initial flight regime), the intensity of the warning buffet is much less. This is caused by the smaller pulsations of pressure in the flow over the relatively short wing, as well as by the higher rigidity of the construction of this wing. An intense buffet occurs when the frequencies of the aerodynamic pulsations (pressure pulsations) and the first harmonic of the bending vibration of the wing coincide.

The frequency of aerodynamic pulsations usually amounts to 3 to 8 oscillations/sec. The frequency of the first harmonic of the bending vibration of the wing in subsonic aircraft is usually within the limits of 5 to 7 oscillations/sec. In supersonic aircraft, the wing is much more rigid, due mainly to its reduced length and more massive construction, selected to ensure the required characteristics of aeroelasticity in flight at very high supersonic Mach numbers and velocity heads. Increase in the rigidity of the wing has led to an increase in the frequency of its natural vibrations. The frequency of the first harmonic of the bending vibration of the wing in supersonic aircraft is about 8 to 11 oscillations/sec.

Figure 3.17 shows an example of the change in the coefficients of force with Mach number. The graph shows clearly the characteristic increase in the difference $\Delta c_{yj} = c_{yst} - c_{yj}$ with increased Mach number. At low (near-zero) Mach numbers, the value Δc_{yj}, as we have already pointed out, is usually very small or even equal to zero. At near-sonic speeds, Δc_{yj} can reach 20 to 30% of c_{yst}. It is desirable for the value Δc_{yj} not to be equal to zero (to form in the aircraft a natural system of signaling the pilot that the aircraft was approaching a near-critical angle of attack), and also that it not be too large (the latter leads to an excessive drop in /91

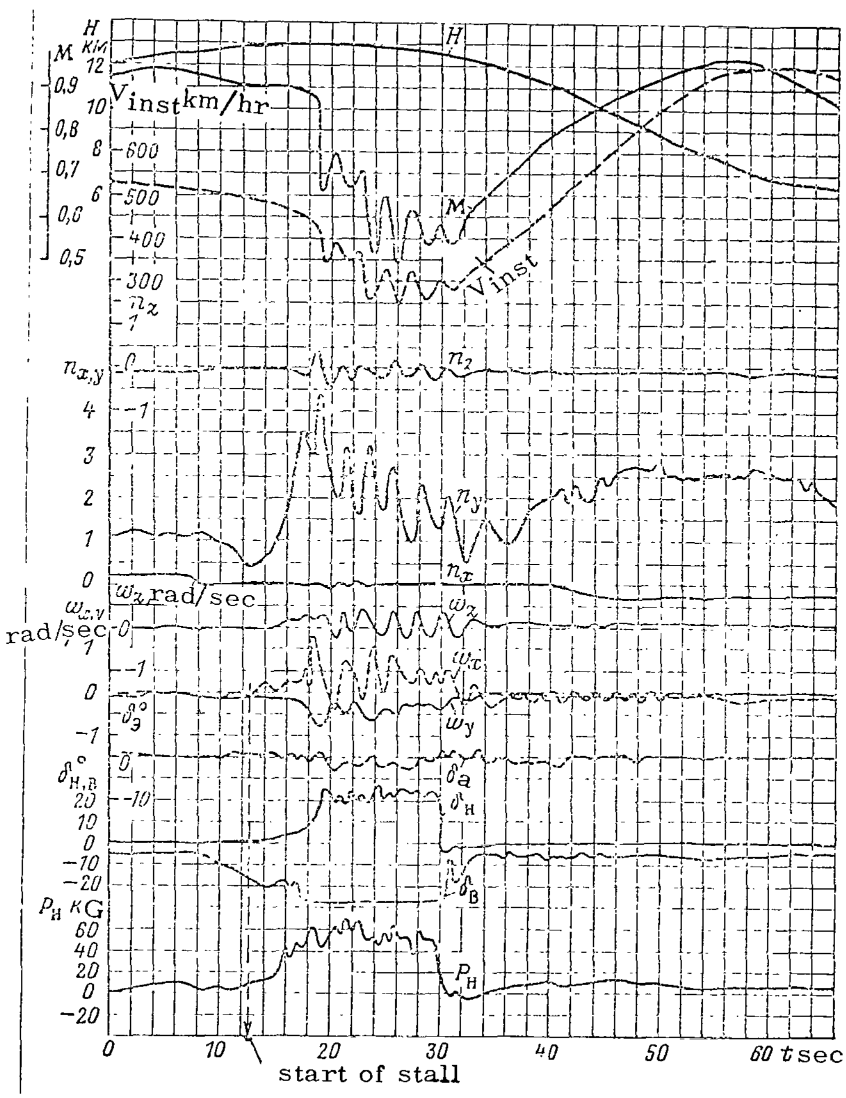

Fig. 3.16: Stalling of an Aircraft at High Instrument Flight Speed During Vertical Maneuver.

the range of maneuverability of the aircraft in the course of its normal operation). /92

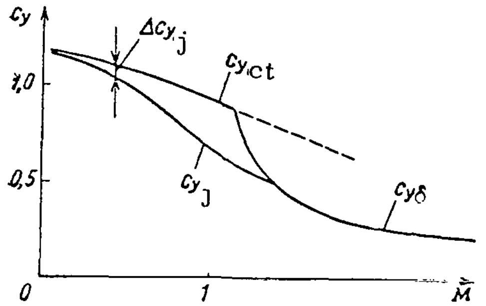

Fig. 3.17: Example of the Dependence of the Coefficient of Lift upon the Mach Number in Supersonic Aircraft.

The values $c_{y_{st}}$ and especially c_{y_j} themselves decrease considerably when the Mach number increases. Thus, for example, when the Mach number increases from near-zero to near-sonic values, the value of $c_{y_{st}}$ can decrease by 25 to 35%. A decrease in c_{y_j} is usually still more considerable. As the Mach number increases due to the decrease in effectiveness of the longitudinal control, the values of the coefficient of lift also decrease, when the aircraft can be stabilized (at moment M_z) with full deflection of the control stick backward; this is the maximum value of c_y which can be attained at supersonic flight speeds, usually designated by the subscript "δ": c_{y_δ}. The value of c_{y_δ} depends on the flight alignment of the aircraft: a shift of the center of gravity forward (increase of the degree of longitudinal static stability of the aircraft under stress) leads to a decrease in c_{y_δ}, and vice versa. /93

(c) STALLING IN UNCOORDINATED MANEUVERS

The most abrupt and non-uniform variety of stall is that which occurs at high instrument speeds and initial sideslip (especially with rotation) of the aircraft; this is the third group of stalling regimes. In this case, besides the influence of the high instrument speed and Mach number, important roles are also played by the asymmetry, inequality and instability of the flow over the aircraft in the initial regime (at $\beta_0 \neq 0$, $\omega_{x0} \neq 0$, $\omega_{y0} \neq 0$). Frequently, during a stall in an improper (uncoordinated) banking maneuver (Fig. 3.18), because of the drop in the resultant aerodynamic force, its vertical component also decreases. As a result (due to the existence of an initial roll angle $\gamma_0 \neq 0$) in the case of stalling, for example, in a left bank), a sideslip develops toward the left (inner) wing, which leads to a reduction of the effective sweepback angle of that wing and consequently to a certain increase in its lift.

The inward sideslip produces an increasingly intense stalling on the outer wing (its effective sweepback angle increases). All of this leads to a change in the direction of rotation and a sudden stalling of the aircraft, so that it sideslips toward the outer wing (in this case, the right wing). Thus, stalling during a banking maneuver as a result of moving the control stick too far has a sharply pronounced oscillatory character and involves a change in the direction of rotation of the aircraft. The direction of the spin which then occurs is opposite in direction to the rotation of the aircraft in the banking maneuver. Thus, as we can see from Figure 3.18, it is often the case that the stall was preceded by a left turn as the aircraft was making a left banking maneuver (ω_y > 0 according to the accepted rule of signs). After stalling began, the direction of rotation of the aircraft changed and the latter went into a right-hand spin (ω_y < 0, the aircraft turning its nose to the right). In the example which we have been describing (see

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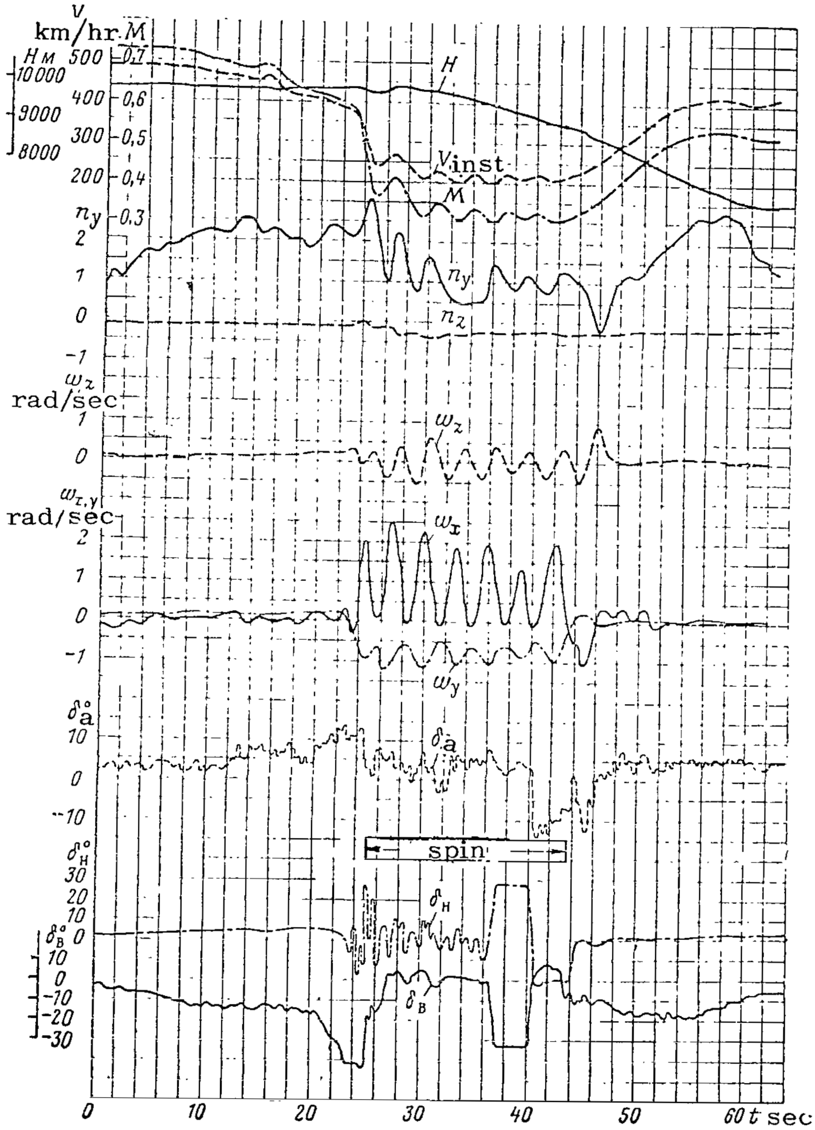

Fig. 3.18: Stalling of an Aircraft During a Left Bank (t = 23 sec) and Subsequent Transition to a Right Normal Spin (t = 25 sec).

Fig. 3.18), the gyroscopic moment of the turbojet rotor (rotation of the engine rotor to the left) with $\omega_z > 0$ tended to force the nose of the aircraft toward the left during the stall, i.e., it promoted the occurrence of a sideslip to the right (outward).

(d) CHANGES IN THE ANGULAR VELOCITY AND ROLL ANGLE DURING A STALL.

Changes in the angular velocity of rolling with time during stalling of an aircraft are of two main types:

(1) With variations of the angular velocity of the roll at which (in the process of stalling) not only its absolute value changes but even the sign as well (Fig. 3.19,a). /95

(2) Practically without variations or with variations of the angular roll velocity at which the sign does not change (see Fig. 3.19,b).

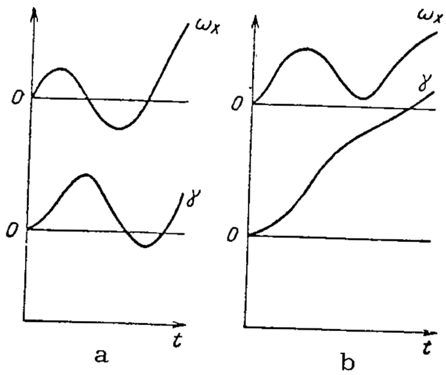

Fig. 3.19: Two Types of Change in Angular Velocity and Roll Angles in the Stalling of an Aircraft.

In the first case, there can also be a change in the sign of the roll angle (the aircraft "rocks" from one wing to the other). As mentioned above, a picture of this kind is usually observed during stalling of an aircraft in random maneuvers in space, particularly in the presence of an initial roll or sideslip ($\gamma_0 \neq 0$, $\beta_0 \neq 0$, $n_{y0} > 1$). In the second case, the roll angle during the stall increases steadily, and the aircraft rolls continuously from one side to the other, right from the start of the stall. This corresponds to stalling from loss of speed.

3.4. Stalling at High Supersonic Speeds and at the Dynamic Ceiling

(a) STALLING AT SUPERSONIC MACH NUMBERS

Regardless of the impossibility of the aircraft's attaining $c_{y\,st}$ at high supersonic flight speeds, even in the case of a complete deflection of the control stick backward when carrying out coordinated maneuvers ($c_{y\delta} \ll c_{y\,st}$, see Fig. 3.17), it is still possible for the aircraft to stall at Mach numbers greater than 1. The fact that an aircraft can stall at high super-sonic Mach numbers is due primarily to two causes. In the first place, the decrease in the degree of static directional stability of the aircraft at M >> 1 means that large sideslip angles can develop, as a result of which (due to the appearance of unsymmetric flow) stalling can occur at values of c_y which are much smaller than $c_{y\,st}$ in the case where $\beta = 0$. An example of an aircraft stalling in such a regime is given in Figure 3.20. In this case, the stall which occurred as the result of the asymptotic increase in sideslip ($n_{z\,max}$ 1.6 at time $t \approx 19$ sec in Fig. 3.20) produced by the loss of directional stability, occurred very abruptly, so that the pilot did not succeed in overcoming the subsequent transition of the aircraft into a spin.

In the second place, at high supersonic Mach numbers, due to the decrease in the degree of static directional stability of the aircraft, it is possible to have abrupt occurrence of an interaction between the longitudinal and lateral motions of the aircraft (see Chapter I). As a result of the aerodynamic and inertial interaction of the motions, there may arise unintentional stresses when the air- /96

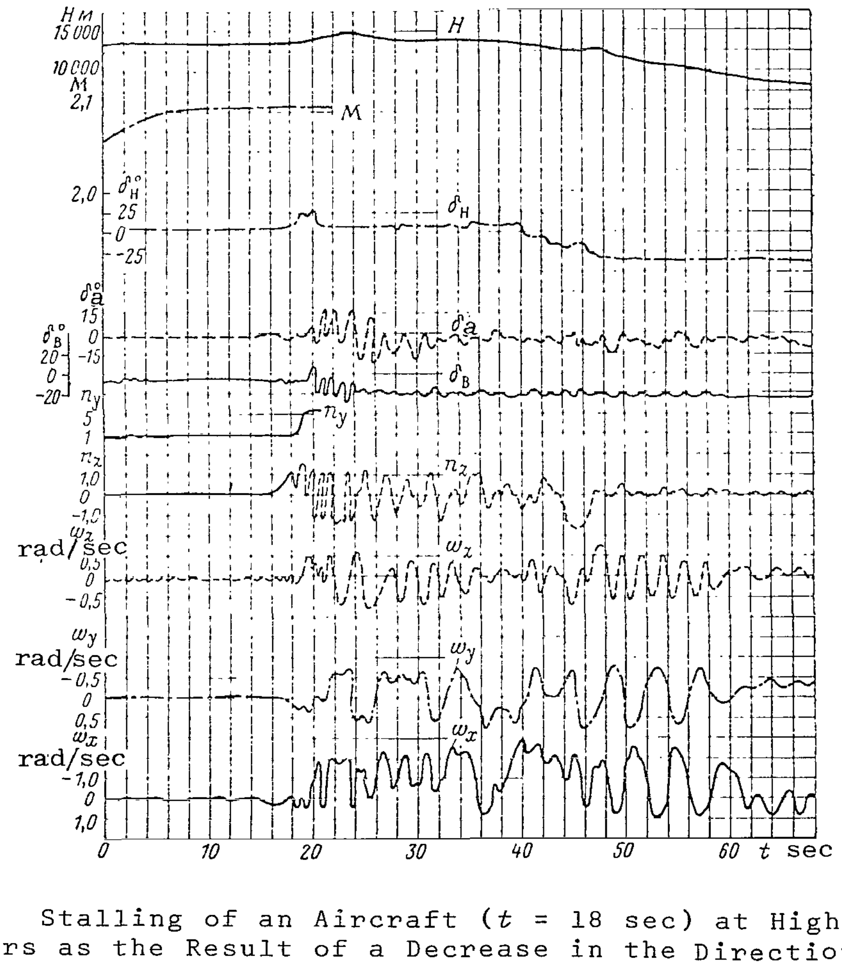

Fig. 3.20: Stalling of an Aircraft (t = 18 sec) at High Supersonic Mach Numbers as the Result of a Decrease in the Directional Stability and Subsequent Transition to a Spin.

craft attain $c_{y\,st}$. An example of this situation in flight is shown in Figure 3.21. It is clear from the graph that the stall was accompanied by the appearance of high absolute vertical and lateral), and by marked oscillation of the aircraft.

(b) STALLING AT THE DYNAMIC CEILING /97

Flight regimes at the dynamic ceiling have great significance and special characteristics in supersonic aircraft. As a rule, there are no basic differences between the behavior of the aircraft in a stall at the dynamic ceiling and a stall at high (but not dynamic) altitudes (not above the static ceiling). However, the predisposition of an aircraft to stall is greater at the dynamic ceiling, due mainly to the considerable drop in flight speed which occurs and to the decrease in the degree of aerodynamic damping (see Chapter I).

With approach to the stalling regime at the dynamic ceiling, the vibrations of the aircraft (especially its oscillation from wing to wing) increase markedly and begin at higher instrument flight speeds, but the stall itself usually proceeds more sluggishly. This is because the aerodynamic moments of autorotation, as well as the aerodynamic moments of roll damping at subcritical angles of attack, are proportional to the density of the air. Hence, the stalling of an aircraft at lower altitudes (under nearly the same conditions) usually takes place much more abruptly. An example of the stalling of an aircraft at the dynamic ceiling is shown in Figure 3.22. It is particularly clear from the graph that the stalling of the aircraft occurred at a velocity $V_{inst} \approx 200$ km/hr, while transverse oscillations of the aircraft had already occurred prior to stalling, at a velocity of $V_{inst} \approx 500$ km/hr.

Stalling at the dynamic ceiling, as at any other altitude, can send the aircraft into a spin. In this case, the aircraft can be pulled out of the spin only at much lower altitudes (relative to the altitude where the stall began), since the aerodynamic surfaces are relatively ineffective at dynamic altitudes. This fact is not dangerous from the standpoint of the altitude reserve. However, such a prolonged period of spin at high altitudes means that the pilot must be exposed for a long period of time to a state of continuous and highly unsteady rotation of the aircraft (about which more will be said later on), thus making it harder for him to work and orient himself, and it may make him feel ill. This means that the pilot must operate very skillfully when flying at the dynamic ceiling.

Flying practice and analysis of flight accidents shows that stalling of aircraft at dynamic altitudes can occur for two basic reasons:

(1) Instability of the aircraft under force at high angles of attack.

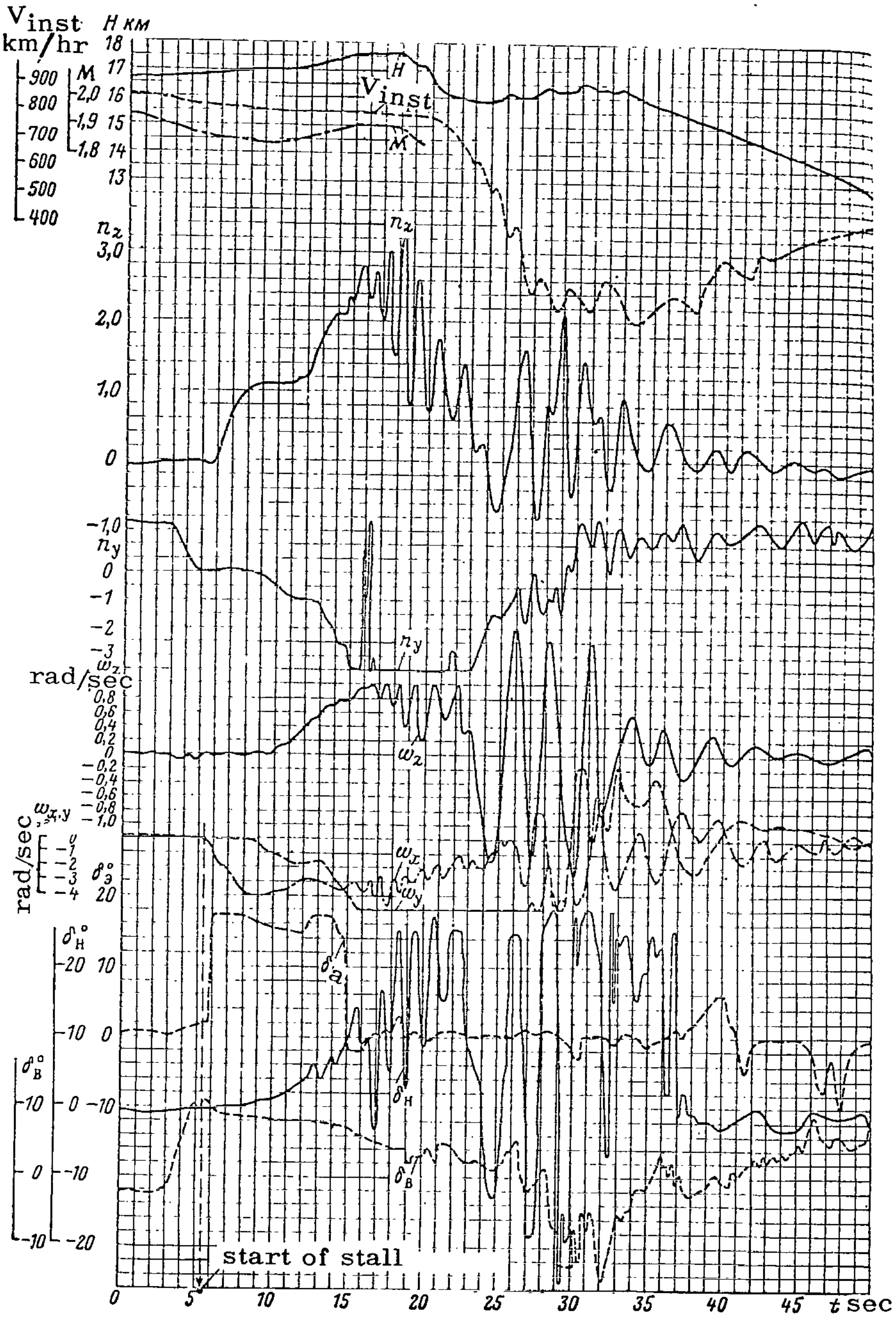

Fig. 3.21: Stall at Supersonic Speed under the Influence of an Interaction Between the Longitudinal and Lateral Motions of the Aircraft, with Subsequent Transition to Spin.

/98

(2) Appearance of yaw, caused by the action of the gyroscopic moment of the engine rotor when the angular velocity of pitch is attained.

(3) Loss of stability by the aircraft under the influence of inertial and (to a lesser degree) aerodynamic interaction of its longitudinal and lateral motions. /99

The possibility that an aircraft will reach dynamic altitudes is related to the conversion of kinetic energy, which it has stored by acceleration at lower altitudes, into potential energy; i.e., there is an unavoidable loss of flight speed when climbing vertically. Consequently, flight at the dynamic ceiling and altitudes near it, where the aircraft possesses a large amount of potential energy accumulated by consumption of a large part of its kinetic energy, occurs at very low instrument speeds and reference pressures.

Low velocity heads reduce the effectiveness of aerodynamic control devices and cause deterioration of the stability characteristics of the aircraft. Hence, in order to increase the safety of flight at these altitudes, it is necessary to fly at very small (in absolute value) angles of attack so that the normal load factor will usually be close to or even equal to zero (see, for example, Fig. 3.22, $t \approx 330$ to 350 sec). Therefore, the absolute values of the aerodynamic forces and moments acting on the aircraft under such flight conditions are small, but this means that their effect on the flight dynamics will also be much weaker than at static altitudes (not above the static ceiling). The latter is an additional argument that flight at dynamic altitudes, especially when climbing vertically, must be performed at small angles of attack: an increase of the aerodynamic lift of the aircraft with an increase in the angle of attack has practically no effect on the nature of the flight trajectory, which essentially forms a ballistic curve, i.e, it gives practically no advantage with respect to the value of the dynamic ceiling, and can only promote creation of a stall.

An analysis of data from flights at the dynamic ceiling and the results of analog modeling of such regimes on electronic computers shows that in this case the inertial forces and moments prevail over the aerodynamic ones. However, owing to the low angular velocities of the aircraft's rotation in flight at altitudes equal to or close to the dynamic ceiling (in the upper part of the climb), the effect of these forces and moments of inertia on the flight dynamics can still be only secondary relative to the effect of the gyroscopic effect created by the moving parts of the engine.

Therefore, the physical picture of the stalling of an aircraft at dynamic altitudes usually resembles the following: As we have already seen, the small size of the aerodynamic moments at such altitudes make it possible to attain only relatively low angular velocities of rotation in the aircraft. However, even the formation of such an angular velocity of pitch can lead to the appearance /10 of relatively large gyroscopic moments of yaw. These moments are produced by relatively large polar moments of inertia in the turbo-

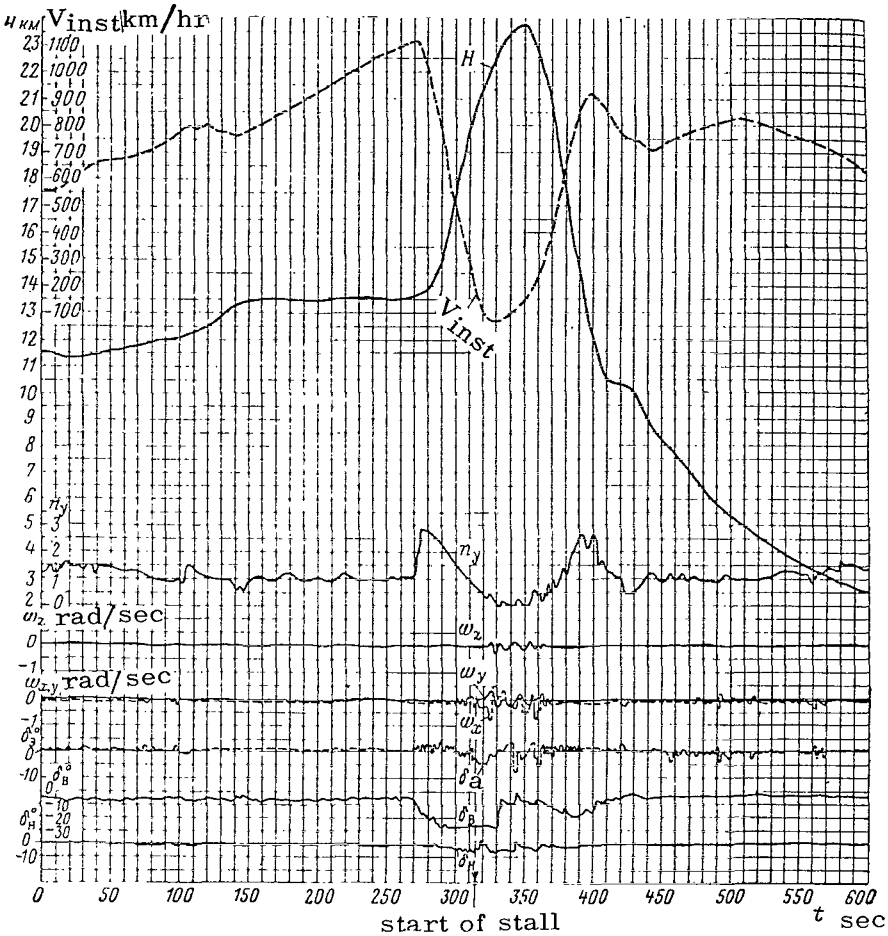

Fig. 3.22: Stall at the Dynamic Ceiling.

jet rotors of modern supersonic aircraft, and the high revolution rate of engines at such altitudes, even when operating them in low-gas situations and with autorotation, they cause the appearance of angular velocities of yaw such that there may be a rather strong inertial interaction between the longitudinal and lateral motions of the aircraft. The latter can also lead to the aircraft going into a stall. /101

Factors which promote stalling also include sideslip caused by the gyroscopic moment of yaw, and poor damping of the free longitudinal and lateral oscillations of the aircraft at high altitudes where the aircraft, to use the pilots' picturesque expression, "doesn't sit right in the air." Any external influence or a more or less vigorous movement of the control surfaces at these altitudes leads to persistent oscillations of the aircraft which occur at relatively high amplitudes.

The initial angular velocities of pitch, which serve as the original cause of such a motion of the aircraft, may be induced by a sudden movement of the control stick. If an aircraft is subjected to unstable load factor at large angles of attack, and if it achieves these angles of attack as a result of an impulse given to the elevator, unintentional pitching may occur. Motion of this type may occur quite rapidly and at dynamic altitudes, with the additional effect of the gyroscopic and inertial moments (even at relatively low initial rotation of the aircraft).

Here is one example of stalling at the dynamic ceiling which led to the aircraft's going into a spin at these altitudes.[1] A Student by the name of Smith, at the Test Pilot School at Edwards Air Force Base in the United States, flying an NF-104A (a modified USAF Lockheed F-104A "Starfighter" intended for training astronauts under conditions of weightlessness), was performing a scheduled training flight at the dynamic ceiling. He had to gain an altitude of 27,450 meters while climbing steeply. At an altitude of 25,300 meters (near the top of his climb), a sudden movement of the control stick away from him created a negative angular velocity of pitch and the nose of the aircraft began to rise smoothly. At this time, the true speed of the flight was 408 km/hr, while the instrument speed and the velocity head were very small and amounted to 89 km/hr and 39 kG/m^2, respectively. The jet engine of the aircraft was operating at a low-gas regime, but rotating at high speed (4000 rpm) due to the great flight speed.

An analysis of the flight recorder charts and the data from modeling the conditions of this flight revealed the following. Under the influence of the gyroscopic moment of yaw after the negative angular velocity of pitch was established, the nose of the aircraft began turning to the left and the aircraft started sideslipping to the right. Due to the lateral static stability of the aircraft, this sideslip led to the creation of an aerodynamic moment /102
of roll, so that the aircraft began to roll over onto its left wing, thus changing the angle of attack.

Such a combination of the effects of the gyroscopic and aerodynamic moments produced a complex movement of the aircraft in space, with rotation relative to all three axes. The appearance of angular velocities led to the development of inertial destabilizing moments of yaw and pitch, thus permitting further increase in the absolute values of the angles of attack and sideslip. As a result, the aircraft lost its stability and acquired a larger angle of

[1] See for example: Polve, James H.: Cause Determined; Aerospace Safety. United States Air Force, No. 4, pp. 10-11, April, 1965.

attack; a stall ensued, followed by a spin, from which the pilot was unable to rescue the aircraft. At an altitude of 12 km the pilot ejected himself from the spinning aircraft.

An analysis of the data from such flights on the NF-104A and the testimony of the test pilots shows that the optimum flight path angle to reach the dynamic ceiling in this aircraft is 65 to 70°. Climbing at this angle is performed with an angle of attack of the aircraft on the order of 5°. The angle of the pilot's seat back to the vertical axis of the aircraft is 14°. Consequently, during the climb the body of the pilot is inclined at an angle of 84-89° to the vertical, i.e., the pilot is lying practically horizontal. This unusual and uncomfortable position makes the pilot's work more difficult and makes it impossible for him to pilot the aircraft visually by depriving him of the chance to orient himself with respect to the ground and the true horizon. Hence, maneuvers of this kind are performed on instruments, thus adding additional complications to the pilot's task. This factor indirectly increases the likelihood that the aircraft will unintentionally enter critical regimes. To avoid stalling during training flights when climbing in the NF-104A, the above-mentioned Test Pilot School at Edwards AFB in the USA placed the following limitations on the characteristics of the trajectory: the speed at the peak of the climb must remain supersonic, so that the velocity head will never drop below about 100 kG/m^2, while the maximum climb angle while gaining altitude must not exceed 45 to 50°.

The increase in the initial velocity (Mach number) during stalling at the dynamic ceiling has the same effect as in stalling at middle altitudes: as $V_0(M_0)$ increases, the intensity of the stall and the absolute values of the angular velocities and accelerations ω_x and ω_y, ε_x and ε_y, also increase.

(c) LIMITS OF FLIGHT SPEED ATTAINABLE IN STALLS

In principle, stalling of an aircraft is possible at any velocity when the proper stresses are present. The greater the speed, the greater the vertical load factor during stall, n_{st}. For example, if in an initial regime of rectilinear horizontal flight, (n_{y0} = 1), a stall occurs at an instrument (or rather, indicator) speed of V_{st} = 200 km/hr (n_{st} = 1), then at a speed of 300 km/hr the stall will occur at a load factor of $n_{st} = 300^2/200^2 = 2.25$ (if the effect of compressibility is not taken into account). When the actual vertical load factor n_y is less than n_{st}, then even if the flight speed V becomes less than the stalling speed of the aircraft in a regime of rectilinear horizontal flight V_{st}, the aircraft will not reach α_{st}, i.e., stalling will not occur. This fact is used particularly in flights at the dynamic ceiling, when the aircraft is usually braked to a final velocity much less than V_{st}. /103

For an anlysis of the nature of the change in vertical load factor during flight at the dynamic ceiling, let us use a schematic trajectory of such a flight, as shown in Figure 3.23. In this case, at the end of acceleration (required for the aircraft to acquire the necessary amount of kinetic energy at $n_y = 1$, the pilot pulls the control stick toward him and creates the load factor $n_y > 1$, i.e., he shifts the aircraft into a climb. At the same time, the flight speed begins to decrease. During the process of gaining altitude, the vertical load factor gradually decreases and becomes equal to unity (Point 3 on the figure). Then the vertical load factor continues to drop, and the slope of the flight trajectory begins to decrease. However, thanks to the positive angle of climb of the trajectory $\theta \gg 0$, acquired between Points 2 and 3, the aircraft still continued to gain altitude. At the peak of the trajectory (Point 4) the flight speed $V \ll V_{st}$, but since the true vertical load factor n_y is then less than unity, the aircraft does not stall, if the condition /104

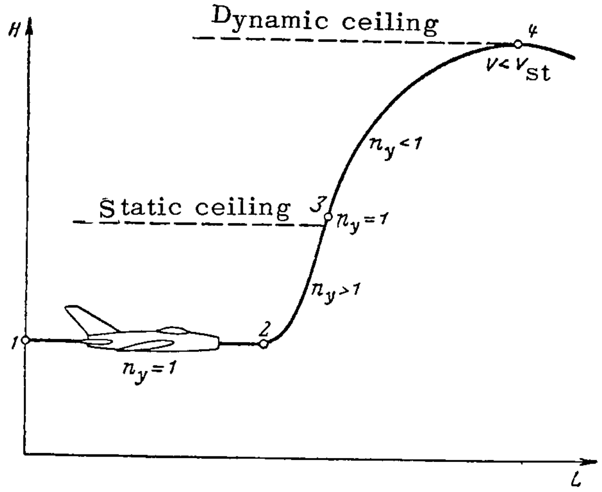

Fig. 3.23: Diagram of the Trajectory of Flight at the Dynamic Ceiling

$$\frac{c_y}{c_{y\,st}} = n_y \frac{V_{st}^2}{V^2} < 1, \text{ i.e. } n_y < \left(\frac{V}{V_{st}}\right)^2 \quad (3.1)$$

is satisfied. The stalling speed of an aircraft in a maneuver $V_{s.m.}$, i.e., the flight speed at which stalling occurs when performing a maneuver with a load factor of $n_y \neq 1$, is determined by the following familiar relationship:

$$V_{s.m.} = V_{st} \sqrt{n_{st}} \quad (3.2)$$

and

$$n_{st} = \frac{c_{y\,st}}{c_{y\,h.f.}} \quad (3.3)$$

Here, $c_{y\,h.f.}$ is the coefficient of lift of the aircraft in rectilinear horizontal flight at a speed of $V_{s.m.}$ The value of $c_{y\,st}$ used here is determined from a graph like that in Figure 3.17, at a Mach number which corresponds to the speed $V_{s.m.}$ at the flying altitude in question.

An example of the limits of flight speeds attainable from stall conditions, obtained according to (3.2) as a function of the force at one specific flight altitude, is shown in Figure 3.24. It is clear from the graph that even at flight speeds close to zero, stalling does not occur if the load factor n_y is sufficiently low and satisfies (3.1). Therefore, in flying at the dynamic ceiling the aircraft can brake to very slow speeds. However, in order for the aircraft to remain maneuverable at this time, it is necessary to limit the given flight speed (velocity head) at the dynamic ceiling, at which the aerodynamic surfaces are still sufficiently effective. This enables the pilot to use these surfaces to overcome any influences that may tend to divert the aircraft from the desired flight regime. /105

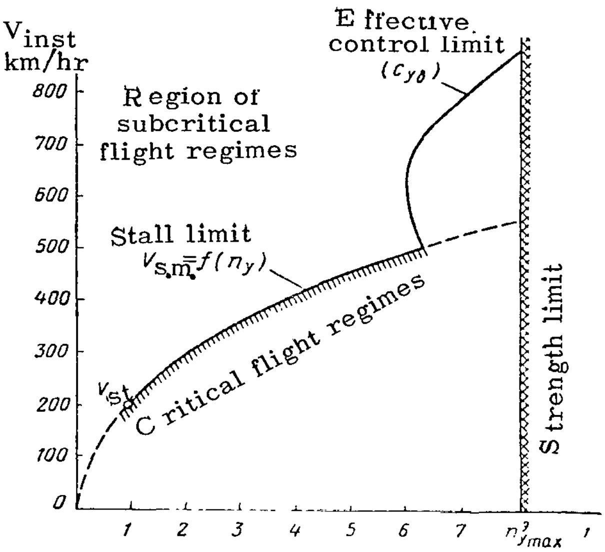

Fig. 3.24: Limit of Flight Speeds Attainable from Stalling Conditions While Performing Maneuvers at a Prescribed Altitude.

In addition to those listed above, there are a number of other factors which can have a significant influence on the characteristics of a stall: the external configuration of the aircraft, the methods of piloting it (especially promptness and consistency in moving the control surfaces), the operation of the engine, the position of the aircraft before its entrance into critical regimes, etc. Therefore, stall regimes [even for one aircraft under identical initial velocities (Mach numbers), flight altitudes and load factors] can be very different.

3.5. Stalling Due to Turbulence

The practical danger of an aircraft attaining α_{st} when it encounters turbulence comes from vertical air currents. In other words, when an aircraft is flying in turbulent air, only the vertical components of the velocities of shifting air masses are generally significant. Horizontal currents or disruptions of wind flow are of insignificant influence in changing the angle of attack of the aircraft, and therefore can lead to a stall only during flight in peripheral regimes (at speeds sufficiently close to stalling speed). However, it is not very likely that the pilot would be flying in such regimes when he encountered turbulence.

Let us examine the conditions for an aircraft's encountering a vertically rising current of air. If the angle of attack of the aircraft was α_1 before it encountered this current, then after the aircraft enters this rising current, which is moving at a speed W, its angle of attack will increase by the value $\Delta\alpha$ and will be equal to $\alpha_2 = \alpha_1 + \Delta\alpha$(Fig. 3.25). The value $\Delta\alpha$ is determined from the expression

$$\Delta\alpha = (\tan^{-1} \frac{W}{V}.$$

In order to place the aircraft in a stall regime, the rising current must undergo an increase of the angle of attack $\Delta\alpha \geq \Delta\alpha_{st}$, where $\Delta\alpha_{st} = \alpha_{st} - \alpha_1$. This means that its vertical speed must be

$$W_{st} = V \tan\ (\Delta\alpha_{st}) \quad (3.4)$$

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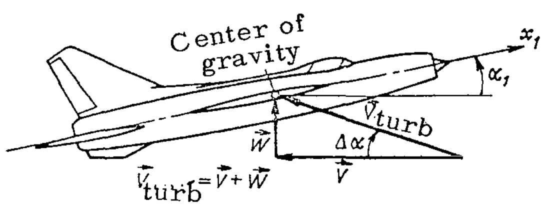

Fig. 3.25: Change in the Angle of Attack When an Aircraft Enters a Vertical Disruption of Wing Flow

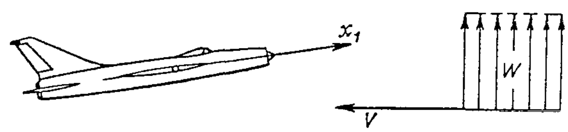

Fig. 3.26: Appearance of Rising Current of Air, Sketched Arbitrarily.

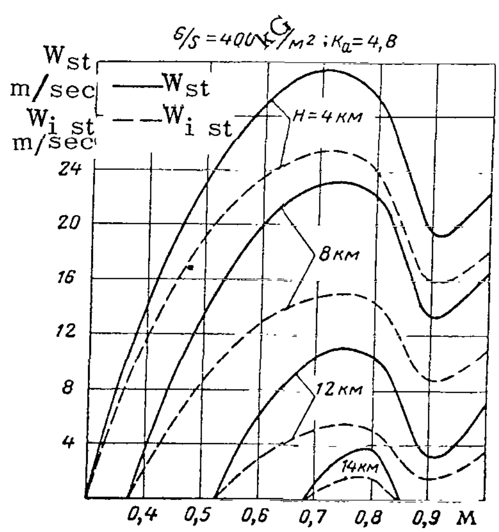

Fig. 3.27: Curves $W_{st} = f(M,H)$ and $Wi_{st} = F(M,H)$.

In addition, the rising current must have sharply defined limits (Fig. 3.26), and the aircraft must strike it instantaneously. /107
Although the latter two conditions are not fulfilled in practice, they serve to make our considerations easier without changing appreciably the conclusions which are drawn.

It is clear from Figure 3.24 that the smaller the angle of attack of the aircraft (i.e., the larger $\Delta\alpha_{st}$) and the greater the flight speed, the greater must be the speed of the rising air which is capable of throwing an aircraft into a critical regime. However, as the flight speed (i.e., the Mach number) increases under the effect of air compression, the value α_{st} can decrease significantly. Consequently, the value $\Delta\alpha_{st}$ can decrease more than the flight speed increases. Therefore, the speed of the rising current W_{st} at certain Mach numbers can also decrease with an increase in the Mach number (speed) of flight.

Consequently, for each flight altitude of the aircraft, there is a certain range of Mach numbers at which the value W_{st} will be maximum. In Figure 3.27, there is an example of the dependence of the values of the true (W_{st}) and indicated (W_{ist}) speeds of the vertical updrafts in the wind, upon the Mach number and flight altitude. The values W_{st} and W_{ist} are linked by the familiar equation $W_{st} = W_{ist}/\sqrt{\Delta}$, where Δ is the relative density of the air at the flight altitude in question. It is clear from the graph that as the flight altitude increases, the values W_{st} and especially W_{ist} decrease markedly. The shapes of these curves for a given aircraft must always be taken into account when selecting the optimum values for the Mach number and flight altitude which will ensure maximum safety of flight in case of severe turbulence.

3.6. Prevention of Stalling and Recovery from It.

Stalling of an aircraft, which is usually unexpected by the pilot, is always dangerous since it is accompanied by partial and sometimes even complete loss of maneuverability of the aircraft. After stalling, the aircraft may start spiraling or go into a spin. In the latter, the maneuverability of the aircraft is much worse than under normal operational flight regimes; as a rule, however, the pilot can pull the aircraft out of the spin by careful piloting. In a spiral, the maneuverability of the aircraft returns almost completely after a while if the pilot does not make serious errors in controlling the aircraft during this time. In both cases mentioned (especially in the case of going into a spin), there is considerable loss of altitude, which is especially dangerous when the aircraft is originally at a low altitude. /108

If the stall has occurred at high speed, the maximum operational stresses may be exceeded (appearance of residual deformations or even fractures in the structure); likewise, the extreme attainable Mach numbers and velocity heads may be exceeded when the aircraft goes into a spiral. Stalling which occurs near the ground, (for example, when coming in for a landing) means that the aircraft falls with considerable vertical speed. This is usually accompanied by a very unsatisfactory attitude of the aircraft immediately before landing, so that it becomes difficult to bring it down on the landing gear. In such cases, it is often the case that the aircraft will touch down when it is tilted to one side, so that one wingtip strikes the ground and causes an accident.

In this connection, we should mention one characteristic and very dangerous piloting error: an attempt to halt the descent of an aircraft moving along a spiral trajectory after going into a stall, by pulling the control stick backward. This only makes the rotation of the aircraft more intense and the speed of descent increases instead of decreasing. The likelihood of committing such an error is increased still further by the fact that in a supersonic aircraft, rotation when moving along such a spiral is less noticeable to the pilot (due to less angular velocity of rotation of the aircraft) than was the case in old subsonic aircraft.

Therefore, from the viewpoint of ensuring the safety of flight, an important role is played by the presence in the aircraft of natural or specially introduced signals which will clearly warn the pilot in plenty of time that he is approaching a critical regime. Natural warning devices include: warning buffet, pull on the control stick, rocking of the aircraft from side to side, occurrence of yawing motions, etc. If natural signals are not given or are noticed too late, the pilot must be warned of the approach of stalling by special devices for indicating and signaling the aircraft's approach to near-critical regimes. A lack of reliable devices for warning of the approach of critical regimes keeps the pilot from piloting his aircraft reliably and utilizing all of its maneuverability to the full.

(a) DEVICES TO WARN OF STALLING

A modern supersonic aircraft has a great many different limitations which set the limits of safe regimes of flight by instrument speed, Mach number, load factor, angle of attack, regimes of operation of the engine, etc., many of which also depend on changes in the altitude and flying weight of the aircraft. This complicates the pilot's task and makes additional demands on his time and attention for estimating the validity of one or another flight regime. Under /10 certain particularly complex conditions (flying in rough meteorological conditions, pursuing an enemy aircraft in aerial combat, etc.), there may not be adequate time and attention left to the pilot for carrying out these additional tasks. Consequently, there is an increased likelihood that the aircraft will unintentionally enter inadmissible flight regimes and then go into a stall. To avoid this, the aircraft must be fitted with reliable devices to warn the pilot of the approach of disruptive regimes.

The weaker and later appearance (and occasional absence) of warning jolting and a significant change in the values $c_{y\,\mathrm{st}}$ and α_{st} in the range of operational speeds and Mach numbers of flight in supersonic aircraft relative to subsonic aircraft has prompted designers to give increasing attention to special signaling devices which will warn the pilot of approach to critical regimes or even automatically steer the aircraft away from these regimes, should the pilot fail to take any action. A warning of this kind can be delivered by two interacting parameters: the magnitude of the angle of attack, or the coefficient of lift of the aircraft. In addition, due to the significant influence of sideslip on the nature of the curve $c_y = f(\alpha)$ in the region of near-critical angles of attack, it is also useful to coordinate information on the actual values of α or c_y with information on the actual value of the sideslip angle β.

Warning of the approach to near-critical flight regimes must be given to the pilot earlier; for example, at a speed which exceeds the stalling speed by no less than 5 to 7%, or at an angle of attack 2 to 3° less than α_{st}. The magnitude of this margin is selected for each specific type of aircraft on the basis of its characteristics of stability, maneuverability, and special piloting requirements at near-critical regimes, and in some cases also on the characteristics of the initial flight regime, preceding the entry of the aircraft into larger angles of attack (for example, the rate at which the pilot pulls the control stick toward himself, or the rate of increase of the angle of attack of the aircraft). It can be increased when necessary (for example, when the aircraft is unstable under force at high angles of attack), or it can be decreased (for example, with a sufficient degree of longitudinal static stability of the aircraft under force , as well as a sharp increase in the forces and required deflections of the control stick when entering the regime $c_{y\,\mathrm{st}}$). At low altitudes, the limitations on the angle of attack are determined mainly at low flying speeds with values of $c_{y\,\mathrm{per}}$ (stall limit), while at high flying speeds, by the value of the maximum operational stress (strength limit). Usually, at high altitudes over the entire operational range of speeds and Mach numbers, the limitation is set only by the /110

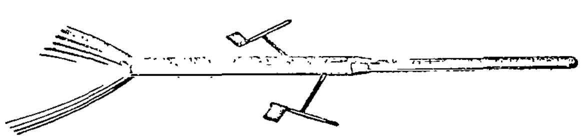

Fig. 3.28: Example of Installation of Vanes for Measuring Angles of Attack and Sideslip Mounted on the Nose of an Aircraft.

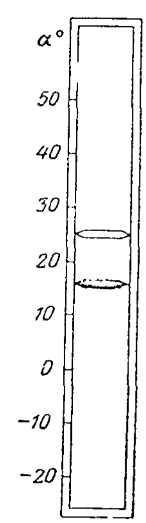

Fig. 3.29: General View of the Scale of the Indicator for Actual (Black Arrow) and Arbitrarily Obtained (White Arrow) Values of the Angle of Attack of an Aircraft.

values $c_{y\,per}$.

Devices for preventing the aircraft from entering critical regimes, based on a determination of the values of the coefficient of lift of the aircraft, have the advantage that all of the special measurement apparatus required for their operation (in particular, sensors for force and fuel consumption, to determine the actual flying weight of the aircraft) is located inside the aircraft. A disadvantage is the need to measure the actual values of the flying speed of the aircraft, which is related to certain difficulties in designing and adjusting these devices. In addition, with approach to near-critical flight regimes, the readings of instruments of this kind become less accurate; due to the presence of the maximum of the curve $c_y = f(\alpha)$, the same values of c_y can occur at subcritical and supercritical flight regimes. Therefore, for example, the readings which the pilot obtains for c_y at near-critical regimes can serve only to disorient him.

At the present time, as a rule, warning devices are used which are based on the measurement of the angle of attack of the aircraft. This is done by installing special vanes in the oncoming flow (Fig. 3.28), whose motions are transmitted to an indicator and signaling device. The most reliable version is the double-vane indicator for the angle of attack. Figure 3.29 shows a general view of the scale of such an indicator. A double-vane indicator of this type shows the actual (α) angle of attack of the aircraft and the value obtained arbitrarily in normal flying operation (α_{per}). By looking at the distance between the two arrows on the indicator, the pilot can get an idea of the reserve of angle of attack available to him for carrying out required maneuvers. When the actual angle of attack increases to the arbitrary limit set for it, additional signals are triggered (audible or visual signals, etc.).

The advantage of such warning devices is the absence of a need to determined the flying weight of the aircraft; the disadvantage is the need to mount vanes in the external flow (there is a chance that they will be damaged or their setting changed while the aircraft is being serviced on the ground). It is also necessary to convert the readings. The latter is due to the fact that the vanes do not actually measure the angle of attack of the aircraft, but the angle between the direction of the local streamline (at the point where the vanes are mounted) and the axis of the aircraft. The difference between this angle and the actual angle of attack of the aircraft (i.e., the correction of the value indicated by the vanes) is not constant. It can change significantly, depending on the flying regime, the operating regime of the engine, the external configuration of the aircraft, etc. /11

If this correction were constant, its value would not be significant; it could easily be taken into account when calibrating the instrument. In those cases where this correction changes significantly as a function of flight conditions, use of the vane-type indicator of the angle of attack is considerably more difficult. It means that when the device is calibrated, the average value of the correction must be taken into account first, and only then must the corresponding corrections be made in reading the instrument for the deviation of the actual value of correction from its average value under various flight conditions. Therefore, the place where the vanes are mounted on the aircraft should be chosen whenever possible so that the values of these deviations will be minimal. The signaling device may also be linked to an automatic device for moving the control stick (independently of the pilot's action) forward in order to attain a certain angle of attack.

(b) RECOVERY FROM STALL

If it is impossible to prevent the aircraft from attaining $c_{y\,st}$ (i.e., a stall occurs), the pilot must begin as soon as possible to pull the aircraft out of this regime. In pulling the aircraft out of a stall or out of a regime of high subcritical (close to α_{st}) angles of attack, the pilot must begin by activating only the longitudinal control system, pushing the control stick vigorously away from him. Using the ailerons to combat rolling of the aircraft at high subcritical angles of attack, especially in a stall, is dangerous since in such cases it can only lead to acceleration of the transition of the aircraft into a spin. This is due mainly to the fact that when an aircraft in a stall reaches supercritical angles of attack, tilting the ailerons against the rotation can only aggravate this rotation instead of combating it (see Chapter V for further details). In addition, in the case of transverse oscillation of the aircraft in the process of stalling, the pilot sometimes is unable to tilt the ailerons in the required direction.

In pulling the aircraft out of a stall, it is recommended that the rudder as well as the ailerons be kept in a neutral (initial /112
balanced) position. The pilot must keep the rudder in this position until he is sure that the aircraft has returned to a normal operational angle of attack $\alpha < \alpha_{per}$ (when warning buffet has ceased, the force is reduced, etc.).

This is because the direction of flight (especially without a booster) is the most roughly controlled factor (the legs are less sensitive than the hands). In a stall, therefore, which is an unusual flight regime calling for considerable additional attention, the pilot can easily "shift his leg" (cause an extreme deflection of the rudder to the side opposite that in which the aircraft is "falling"). This can only change the direction of rotation of the aircraft in the stall instead of ending it. Sometimes this even causes the aircraft to shift from the stall condition into a spin, the opposite rotation direction with respect to that from which the pilot wanted to extract the aircraft after the stall.

In addition, in the case of an abrupt stall, the aircraft can rather rapidly assume a nearly inverted position at negative supercritical angles of attack ($\alpha < \alpha^{*}_{st}$). If in this case the pilot tilts the rudder too far against the roll and turn which develop after the stall, while simultaneously moving the control stick away from him (to combat a stall which has occurred at a positive angle of attack), this tilting of the rudder can cause the aircraft to shift to a regime of inverted spin.

If the aircraft, after emerging from the stall, has entered normal operational angles of attack (necessarily subcritical) in the upside-down position or close to it, there are two ways to get the aircraft back into a normal position relative to the horizon: make a "half roll", or complete the second half of the wingover maneuver. The half roll is usually performed, since the speed of the aircraft is then increased much less than in making the second half of a wingover, in the course of which the speed in some cases can even go beyond existing limitations.

The limitations on flight speed and Mach number, as well as flight altitude, angles of attack and sideslip, etc. for mass flight tests must be selected so as to enable the pilot to perform as fully and safely as possible, the required maneuvers as well as routine piloting tasks.

CHAPTER 4

SPIN

4.1. Physical Nature of Spin

(a) AUTOROTATION OF THE AIRCRAFT

The cause of spin is the aerodynamic autorotation which arises after an aircraft has stalled. In the presence of sideslip, aerodynamic autorotation (in the following, we will refer to it simply as "autorotation") of an aircraft mainly involves only the wing. During sideslip, a significant portion of the aerodynamic moment of autorotation can be contributed by the fuselage as well. Let us see how autorotation of an aircraft comes about. /113

AUTOROTATION OF THE WING

1. PHYSICAL NATURE OF AUTOROTATION

Autorotation produced by the wing arises as follows. It has been mentioned previously that the rotation of a supersonic aircraft, after stalling, takes place relative to an axis close to its longitudinal axis ox_1. For the sake of simplicity (the qualitative aspect of the phenomenon will remain the same), we will assume that the axis of rotation coincides with the longitudinal axis of the aircraft. As we know, in this case the angles of attack of the wing cross sections differ from the angle of attack of the aircraft by the value $\Delta\alpha_{sec} \approx \arctan \frac{\omega_x z_{sec}}{V}$, where z_{sec} is the distance of a given cross section from the axis ox_1.

From Fig. 4.1, we can see that the angles of attack of the cross sections in the rising wing (α_{rise}) increase, while they decrease in the falling wing (α_{fall}). At subcritical flight regimes, such a difference in the angles of attack of the wings from one side to the other produces an increase in the coefficients of normal aerodynamic force on the rising wing $c_{y\,rise}$ and a decrease in these coefficients in the falling wing $c_{y\,fall}$; an aerodynamic moment of roll damping is produced (Fig. 4.2, a). At supercritical initial ($\alpha_{initial}$) angles of attack in the central cross sections of the wing, the opposite is the case: the normal aerodynamic force of the falling wing exceeds the aerodynamic force on the

rising wing (see Fig. 4.2.b). The result is an aerodynamic moment /11
which acts in the rotation direction and tends to increase the angular velocity of the aircraft's rotation (the moment of autorotation, M_{auto}). The increase in the angular velocity, and therefore the change in the angles of attack of the wing cross sections, will continue until the coefficients of normal aerodynamic force on the right and left wings are equal (see Fig. 4.2, c). The rotational moment M_{auto} then vanishes and the wings continue to rotate at a constant angular velocity of roll.

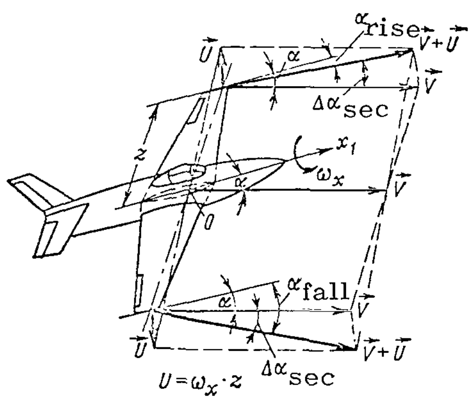

Fig. 4.1. Diagram Showing Changes in Angle of Attack on Wing Cross Sections During Rotation of the Latter Relative to The Longitudinal Axis of an Aircraft.

Such a phenomenon, which occurs at subcritical angles of attack of the central cross sections of the wing, is the result of spontaneous rotation of the wing at an initial angular velocity of roll $\omega_x \neq 0$ after stalling, and is also called autorotation of the wing. Autorotation of the wing can be aggravated still further by an increase in the area of disturbed flow over the rising wing due to the angular velocity of roll $\omega_x \neq 0$, which causes an additional drop in the value cy_{rise}. The autorotational moment of the empennage can be produced similarly. The value of the autorotational moment is determined to a considerable degree by the slope of the curve $c_y = f(\alpha)$ at the point of transition to supercritical angles of attack. The greater and more sudden the drop in the coefficient of lift in the transition through the critical angle of attack, the greater the autorotational moment.

In reality, the rotation of the aircraft does not take place relative to its longitudinal axis, but relative to some axis in space which lies between the axis ox_1 of the aircraft and the direction of the speed vector of flight (see Chapter I). In addition the braking and torsional moments, (strictly speaking) are not produced by the lifting forces of the wing cross sections, but by the projections of the total aerodynamic forces of each wing cross section on a plane perpendicular to the axis of rotation. Under actual conditions, then, besides the autorotational moment which turns the aircraft relative to its longitudinal axis, there is also the autorotational moment as well, which turns the aircraft relative to its normal (vertical) axis. This also contributes to the spiraling of an aircraft in a spin.

In the majority of instances, the aircraft sideslips when it is in a spin. The sideslip motion has a considerable influence on the autorotation of supersonic aircraft. There are two types of sideslip, inward and outward. Inward sideslip occurs when the airflow strikes the aircraft on the side of the inner wing. Outward sideslip occurs when the airflow strikes the aircraft on the side of outer wing. With inward sideslip, the right wing is involved in a right-hand spin and the left wing is involved in a left-hand spin. With outward sideslip, the left wing is involved in a right-hand spin and the right wing is involved in a left-hand spin. Right-hand spin (normal and inverted) means that the aircraft is rotating toward the right, as seen from the cockpit, looking forward. Left-hand spin (normal and inverted) means that the aircraft is rotating to the left. In other words, as observed from the aircraft (looking in the direction of motion of its center of gravity), right-hand normal spin means that the spin is clockwise, while right-hand inverted spin means that the spin is counter-clockwise. In left-hand normal spin, rotation takes place counter-clockwise, while in left-hand inverted spin it is clockwise. /115

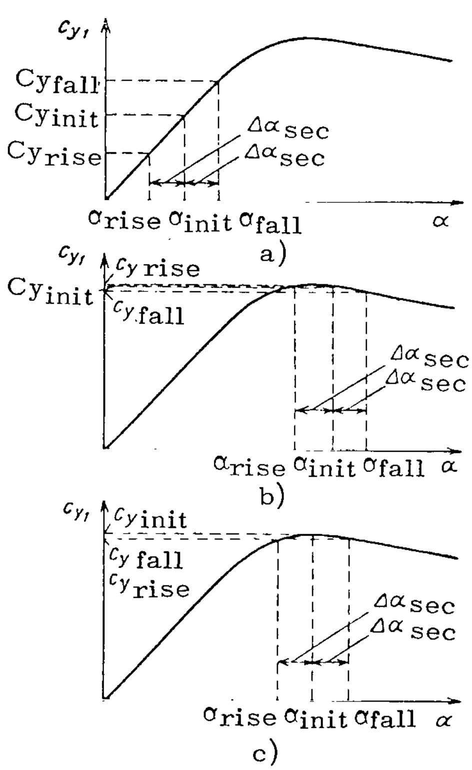

Fig. 4.2. Changes in the Coefficients of Normal Aerodynamic Force on the Cross Sections of a Rotating Wing at Various Initial Angles of Attack.

Sideslip is highly dependent on the characteristics of autorotation of wing. Flight with sideslip involves a shift in the boundary layer and a change in the point of origin of disturbed flow toward the other wing (on which the disruption of flow occurs earlier), thus favoring an earlier development of spin.

An important characteristic is the so-called diagram of autorotation of the wing. A diagram of this sort is obtained by determining the values of the relative angular velocities of roll $\overline{\omega}_x = \frac{\omega_x l}{2V}$, at which the moment of autorotation $M_{auto} = 0$, for various values of the angle of attack at the central cross section of the wing. The curves of autorotation $\overline{\omega}_x = f(\alpha)$ in this diagram show the dependence of the stable angular velocities of autorotation upon the angles of attack of the wing, for various values of the sideslip angle (Fig. 4.3.).

Autorotation of the wing is observed only within a region bounded by the curve $\overline{\omega}_x = f(\alpha)$ and the axis of the abscissa. As we can see from the graph, the area of autorotation expands with outward sideslip and shrinks with inward sideslip. This property of the wing can be used especially in pulling an aircraft out of a spin, when the proper sideslip angle can be set by operating the controls.

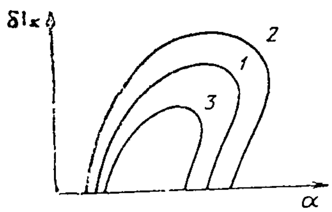

Fig. 4.3. Diagram of Autorotation of the Wing at Various Sideslip Angles: (1) Curve of Autorotation Without Sideslip; (2) Curve of Autorotation with Outward Sideslip; (3) Curve of Autorotation with Inward Sideslip

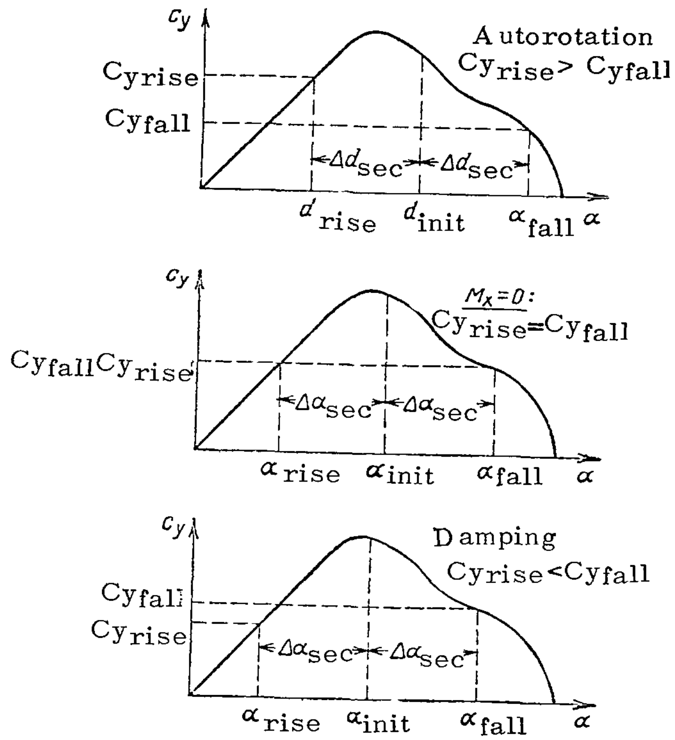

Fig. 4.4. Value and Sign of Aerodynamic Moment of Roll, Produced by a Rotating Wing, versus the Initial Angle of Attack.

/11

At supercritical angles of attack, depending on the shape of the curve $c_y = f(\alpha)$, the combination of the values of the angle of attack at the central cross section of the wing and the angular velocity of its rotation, the existence of sideslip, etc., three things may occur: /11

1. Autorotation of the wing.
2. No rotation of the wing, with a resultant aerodynamic moment of roll equal to zero.
3. Aerodynamic roll damping.

Figure 4.4 shows an example which illustrates the change in value and sign of the aerodynamic moment of roll, generated by a given wing at a fixed angular velocity of roll ($\Delta\alpha_{sec}$ = const), as a function of the value of the initial supercritical angle of attack.

At subcritical angles of attack, the rotating wing can generate only aerodynamic roll damping. Aerodynamic autorotation of the wing cannot arise at subcritical angles of attack. However, as we have pointed out above, the actual value of the critical angle of attack can change considerably as a function of a number of factors (especially the sideslip angle and Mach number).

2. AUTOROTATION OF A SWEPTBACK WING

Autorotation of a sweptback wing, and consequently of an aircraft with sweptback wings, has one significant feature: at relatively low supercritical angles of attack, with a small amount of sideslip, the angular velocity of autorotation of such a wing can undergo considerable changes with time, even leading to changes in the direction of rotation of the aircraft.

For the sake of comparison, we have shown in Fig. 4.5 the autorotation diagrams of two aircraft: one with straight wings ($\chi = 0$) and the other with sweptback wings ($\chi = 45°$). It is particularly clear from the graph that the dimensionless angular velocity of autorotation $\bar{\omega} = \frac{\omega l}{2V}$ in the aircraft with sweptback wings changes sign over a small range of angles of attack with a sideslip angle $|\beta| = 5°$.

It is also apparent from the graph that in the aircraft with sweptback wings, in the absence of sideslip, there is no autorotation at all over a wide range of supercritical angles of attack. When there is sideslip, however, (with uniform absolute values of β), /118
autorotation begins much earlier with a sweptback wing than with a straight wing.

The changes in the direction of autorotation of a sweptback wing at certain combinations of angle of attack and sideslip angle, are related to the nature of the propagation of a zone of disturbed flow over that wing. To illustrate this, Fig. 4.6. shows diagrams of the distribution of areas of disturbed flow over sweptback and straight wings, with increase in the angle of attack during sideslip.

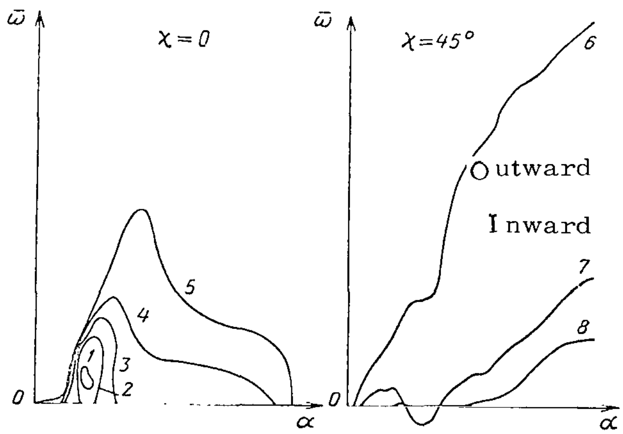

Fig. 4.5. Diagrams of Autorotation of Aircraft with Straight ($\chi = 0$) and sweptback ($\chi = 45°$) Wings. (1) Inward Sideslip $|\beta| = 10°$; (2) Inward Sideslip $|\beta| = 5°$; (3) No Sideslip ($\beta = 0$). (4) Outward Sideslip $|\beta| = 5°$; (5) Outward Sideslip $|\beta| = 30°$; (6) $|\beta| = 30°$; (7) $|\beta| = 5°$; (8) No Sideslip ($\beta = 0°$).

In a sweptback wing during glide, a zone of disrupted flow arises at the end of one wing ($\alpha = 15°$ in Fig. 4.6), which is trailing. As the angle of attack continues to increase, the zone of disrupted flow also appears on the wing which is farther forward ($\alpha = 20°$); as the angle of attack increases, this zone spreads out rapidly and covers the entire wing. At this time, there is still a small area of smooth (non-disrupted) flow on the trailing wing. The result is a change in the direction of rotation of the wing. Later ($\alpha = 40°$ in Fig. 4.6) there is a complete separation of the flow from the wing and the direction of rotation changes again, which is explained by the non-symmetric position of the wing relative to the incident airflow.

For comparison, Fig. 4.6 also shows a diagram of the propagation of an area of disrupted flow across the wingspan of an aircraft with straight wings. It is clear from the diagram that the zone of disrupted flow first arises at the end of the trailing wing and gradually spreads as the angle of attack increases, spreading uniformly to the right over the wings and continuing to occupy an /119
ever increasing portion of them. The direction of the aircraft's rotation does not change.

These differences in the characteristics of autorotation of sweptback wings and straight wings are explained by the following basic facts:

The curve $c_y = f(\alpha)$ has a very flat maximum in sweptback wings. In the absence of sideslip, if the initial angular velocity of the aircraft's rotation is low, the values $c_{y\,\mathrm{rise}}$ and $c_{y\,\mathrm{fall}}$ (see Fig. 4.2,b) will be nearly the same. In this case, autorotation will be less intense, and sometimes will not occur at all ($c_{y\,\mathrm{rise}}$ and $c_{y\,\mathrm{fall}}$ will be practically the same). The picture changes sharply, however, when sideslip is involved. Sideslip means that the effective lengths of the right and left wings of an aircraft with sweptback wings (Fig. 4.7), and therefore their lifting capacities (Fig. 4.8), will be different. The effective length of the forward wing will be

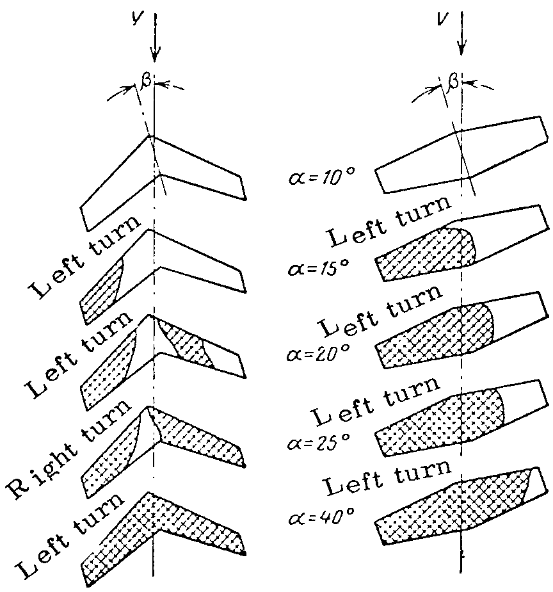

Fig. 4.6. Propagation of Areas of Disrupted Flow over the Upper Surfaces of Sweptback and Trapezoidal Wings, with an Increase in the Angle of Attack During Sideslip (Regions of Disrupted Flow are Shaded).

greater than that of the trailing wing. As the sideslip angle increases, the effective length of the latter decreases, so that at a sufficiently large sideslip angle the trailing wing will have the /120 same effect as if it were shorter. It is clear from Fig. 4.8 that in this case, within the range of angles of attack $\alpha_1 < \alpha < \alpha_2$, autorotation is caused by a relatively high difference between the values of c_y in the left and right wings and is directed toward the leading wing, since its lift is smaller at these angles of attack. A further increase in the angle of attack, when the latter exceeds α_2, means that the lift of the trailing wing will become less. The direction of the autorotational moment then changes.

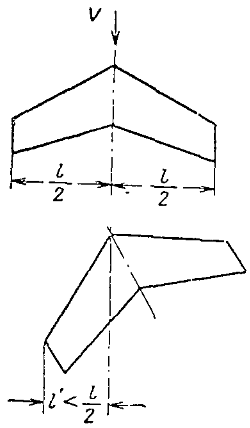

Fig. 4.7. Influence of Sideslip on Effective Length of a Sweptback Wing.

In addition, sideslip entails changes not only in the effective length, but also in the effective sweepback angle of the wings, thus leading to still greater inequality

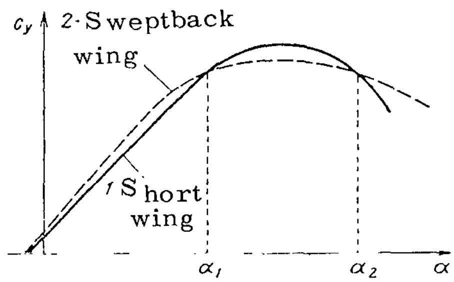

Fig. 4.8. Change of Coefficient of Lift with Angle of Attack in a Short Wing (1) and Sweptback Wing(2).

in the flow characteristics of autorotation. It is clear from Fig. 4.9. that as the effective sweepback angle changes under the influence of sideslip, there is a change in the values of the normal components of the airspeed vector, striking the left wing (normal component $V_{n.l.}$) and the right wing (normal component $V_{n.r.}$). As a result, the effective Mach numbers of the right ($M_{eff.r}$) and left ($M_{eff.l.}$) wings are different. This means that there will also be differences in the aerodynamic characteristics of the two wings. These changes in the lifting capacities of the left and right wings can vary with the value of the initial Mach number of flight and the initial glide angle.

Hence, the characteristics of autorotation for sweptback wings depend to a considerable degree on the combination of the values of the angles of attack and sideslip. Therefore, the entrance of an supersonic aircraft into a spin can occur at still smaller angles of attack than does entrance into a spin for subsonic aircraft with straight wings or wings with a small sweepback angle (the critical angle of attack decreases during a sideslip), regardless of /121 the fact that the critical angles of attack of the latter are much smaller in the absence of sideslip. The above-described nature of the propagation of zones of turbulent flow across the wings of a swept-back-winged aircraft during sideslip can produce periodic variations of the direction of rotation when an aircraft with wings of this type enters a spin. In this case, the transitional portion of the spin may include unintentional transitions from spin in one direction to spin in the other direction, with a fixed setting of the controls during the regime, depending on the spin in the initial direction of rotation.

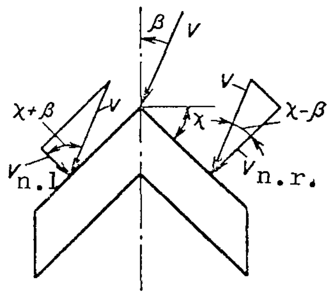

Fig. 4.9. Explanation of the Effect of Glide on the Effective Sweepback Angle of the Wings:

$$V_{n.l.} = V\cos(\chi+\beta);$$
$$V_{n.r.} = V\cos(\chi-\beta);$$
$$M_{eff.l} = \frac{V_{n.l.}}{a} = \frac{V}{a}\cos(\chi+\beta);$$
$$M_{eff.r} = \frac{V_{n.r.}}{a} = \frac{V}{a}\cos(\chi-\beta);$$
$$M_{eff.l} < M_{eff.r}.$$

Of all the types of autorotation, it is aerodynamic autorotation of the wing which usually has the main influence on the characteristics of spin. Hence, the falling of an aircraft into a spin is related to entry to supercritical angles of attack, at which its uninterrupted flow is disturbed and developed regions of disrupted flow appear. As a result of this asymmetric disorientation, reversal and roll moments in autorotation develop (as we have mentioned before) which lead the aircraft to move in a spiral.

AUTOROTATION OF THE FUSELAGE

In some cases, the nose section of the fuselage can be the origin of a relatively large aerodynamic moment of autorotation.

Fig. 4.10 shows an example of the possible relationship and shapes of the curves for the autorotational moment of the fuselage $M_{auto.f.}$ (Curve 1) and the resultant moment of autorotation M_{auto} (Curve 2), as functions of the angular velocity of roll.

Autorotation of the fuselage may arise with certain shapes of the cross section of the nose section, in the event of nonsymmetric flow of incident air over the latter (expecially at large angles of attack).

Studies which have been carried out show that this phenomenon is possible, especially when the shape of the cross section of the nose of the fuselage (Fig. 4.11) is like that shown in Fig. 4.12. If sideslip occurs, the flow in the plane of the transverse cross section of the nose of the fuselage moves at an angle β_f (see Fig. 4.11) to its normal axis oy_1 (Fig. 4.13). It is then possible to have two varieties of position of the normal Y_f and side Z_f /122
components of the resultant aerodynamic force R_f, acting on a given

section of the fuselage: (a) the values Y_f and Z_f are both positive (see Fig. 4.13,a); (b) the value Y_f is positive and Z_f is negative (see Fig. 4.13,b).

Fig. 4.14 shows the nature of the change in the values of Y_f and Z_f with the angle β_f in both of these cases.

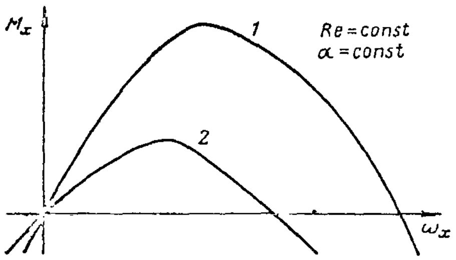

Fig. 4.10. Possible Curve for the Autorotational Moment Produced by an Isolated Fuselage (1) and the Resultant Moment of Autorotation of the Wing-Fuselage System (2).

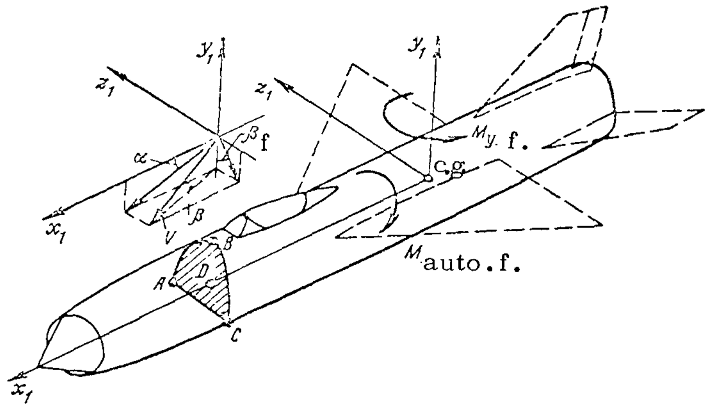

Fig. 4.11. Diagram Showing the Position of the Transverse Cross Section ABC of the Nose of the Fuselage.

Autorotation of the fuselage occurs in the presence of a negative side force Z_f (see Fig. 4.13,b), the point of whose application lies above the longitudinal axis ox_1 passing through the center of gravity of the aircraft (the point of intersection of this axis with the transverse cross section of the fuselage in question is indicated by the letter D in Fig. 4.13). The nature of the flow over the transverse cross section of the nose of the fuselage in this case is shown in Fig. 4.12.

It is clear from the sketch in Fig. 4.12 that the disruption of flow occurs on the upper part of the fuselage, on the side opposite that struck by the incident airflow. This causes a local area of increased pressure (marked by a "+" sign in the diagram). On the other side of the fuselage there is still undisturbed flow, and consequently a region of underpressure (marked by a "-" sign in the diagram).

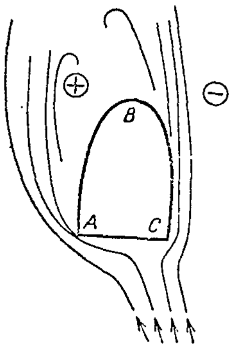

Fig. 4.12. Diagram of Flow over a Transverse Cross Section of the Fuselage, Showing the Reason for Formation of an Autorotational Moment of the Latter.

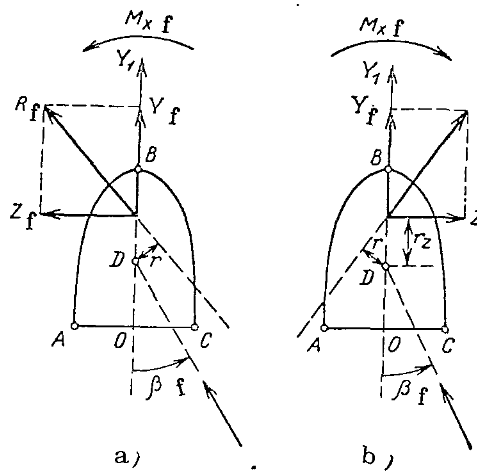

Fig. 4.13. Possible Variations of the Locations of Normal and Side Aerodynamic Forces Acting on the Transverse Cross Section of the Fuselage During Glide. /123

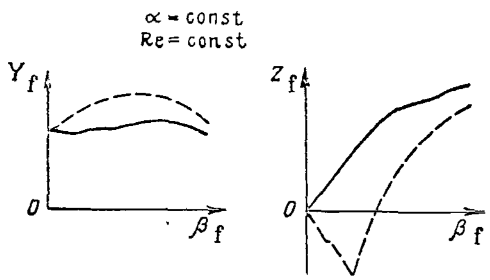

Fig. 4.14. Curves $Y_f = f(\beta_f)$ and $Z_f = F(\beta_f)$ with and without Autorotation of the Fuselage: (—— = Without Autorotation; --- = With Autorotation).

A rise in the underpressure on the side of the incident airflow and a rise in the pressure on the opposite side (in the presence of sideslip, causing nonsymmetric disruption of the flow, is also the cause of the appearance of a moment of the autorotational component in the fuselage. The appearance of an aerodynamic force R_f, applied to the arm r (the distance from the line where this force is applied to the longitudinal axis ox_1) produces the aerodynamic moment of roll and is the autorotational moment $M_{auto.f.} = R_f r$ (see Fig. 4.13). As a rule, the transverse cross section of the aircraft is located in its plane of symmetry y_1ox_1. Then the value $M_{auto.f.}$ is determined solely by the side force Z_f: /124

$$M_{auto.f.} = Z_f r_z$$

The side force Z_f also produces the aerodynamic yawing moment $M_{y.f.} = Z_f l_R$ ($l_R = OD$, see Fig. 4.11). Hence, the autorotation of the fuselage is directly linked to the disruption of its uninterrupted flow and the appearance of a nonsymmetric disruption of flow.

In the absence of sideslip (with $\beta = 0$ and $\beta_f = 0$, see Fig. 4.11), the flow over the right and left halves of the fuselage is symmetric, so that the lateral force is zero. In this situation, autorotation of the fuselage cannot occur.

Consequently, the source of autorotation of the fuselage is the lateral side force which arises in the oose section. Due to the fact that the appearance of the autorotational moment of the fuselage is directly related to disruption of the flow, the value of this moment depends to a considerable extent upon such factors as the shope of the transverse cross section of the fuselage, the roughness of its surface, the magnitude of the angle of attack of the aircraft, and the Reynolds number. The roll angle may also have an important influence on the value of the autorotational moment of the fuselage, since the value β_f changes when the aircraft rotates relative to its longitudinal axis. In isolated cases, the autorotational moment of the fuselage can even arise at subcritical angles of attack, at which the wing always has only aerodynamic roll damping and not autorotation.

INERTIAL AUTOROTATION

An important influence on the spin regime can be exerted by inertial (or rather, aeroinertial) autorotation of the aircraft (see Chapter I). The cause of inertial autorotation is the production of a certain combination of values of the angles of sideslip and attack, the aerodynamic moment of roll, and the inertial destabilizing moment of pitch when the aircraft turns. In this case, a change in the initial angle of glide may be accompanied by the /125 development of an additional angle of roll $\Delta M_{x\beta} = M_x^{\beta}\beta$, acting in the same direction as the aerodynamic moment of autorotation M_{auto}. This change in the initial angle of glide may be the result of both inertial and aerodynamic moments of yaw. The moment $\Delta M_{x\beta}$, producing inertial autorotation of the aircraft, can occur with both supercritical angles of attack and lateral stability (Fig. 4.15), even if lateral instability of the aircraft did occur. In the latter case, only the effect on the angles of attack would be opposite to that described in Chapter I.

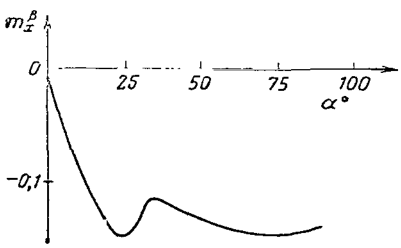

Fig. 4.15. Example of the Change in the Degree of Lateral Static Stability of an Aircraft with Sweptback Wings, as a Function of the Angle of Attack.

The occurrence of inertial autorotation is impossible (by definition) only when the aircraft is neutral with the lateral relationship ($m_x^{\beta} = 0$).

When sideslip is absent, it is impossible for any of the abovementioned forms of autorotation to occur at subcritical angles of attack. With sideslip, the aerodynamic autorotation of the fuselage and the inertial autorotation of the aircraft can occur also at small (definitely subcritical) angles of attack. However, owing to the lack of aerodynamic moments of autorotation of the wing, the occurrence of a characteristic motion of the aircraft along a spiral trajectory (spin) is impossible at subcritical angles of attack. In this case, along with the development of angular velocities of roll and yaw, there are significant aerodynamic damping moments of the wing and empennage (see Fig. 4.2.a), which favor the development of such a motion by the aircraft. The moments of aerodynamic autorotation of the fuselage and the inertial autorotation of the aircraft, superimposed on the moment of the aerodynamic autorotation of the wing, in some cases may have a very significant effect on the spin characteristics. However, these two forms of autorotation (aerodynamic autorotation of the fuselage and inertial autorotation of the aircraft) will not in themselves suffice to produce spin, although the motion of the aircraft when these two factors appear (or even if only one of them appears), even at subcritical angles of attack at the middle sections of the wing, can be very similar to spin. Under such conditions, it is usually difficult for even an experienced pilot to know which regime he is in:a spin or (for example) a regime of inertial autorotation. For a proper understanding of the nature of such regimes, and thus for selection of a method of escaping them, it is advantageous to have on board some visual indicator of the angle of attack. /126

Due to the interaction of longitudinal and lateral motions when an aircraft is in a spin, there may also be an independent change in stress (see Chapter I), which has an important effect on the nature of the regime.

(b) FORCES AND MOMENTS ACTING ON AN AIRCRAFT IN A SPIN

Spin is the result of the superposition of two motions: the relative motion, as exemplified by the rotation of an aircraft around an axis passing through its center of gravity, and a translational motion of the center of gravity itself along a spiral trajectory in space. Since the rotation of a body is determined by the moments acting upon it, and spin is characterized mainly by the rotational motion of the aircraft, it then follows that the features of spin of an aircraft are determined mainly by its moment characteristics. The basic parameters of spin (angle of attack, sideslip angle, and angular velocity of the aircraft's rotation in a regime) depend primarily on the nature of the curves of the moments of yaw and pitch,as well as the moments of roll (to a considerable degree).

DEVELOPMENT OF MOTION FROM STALL TO SPIN

Let us begin by examining the forces and moments acting on an aircraft during development of its motion from stall to spin. We shall assume that in the initial flight regime at subcritical angles of attack, the aircraft was stabilized with respect to its moments, with the elevator in a neutral position. This corresponds to the flight regime shown in Fig. 4.16 by Point 1 (aerodynamic moment of pitch $M_{z\,a} = 0$ at $\delta_e = 0$). The pilot then pulled the control stick all the way back. This produced the aerodynamic moment of pitch, increasing the angle of attack of the aircraft. The aerodynamic moment $M_{z\,a} > 0$ causes the appearance of the angular velocity of pitch $\omega_z > 0$ (with the nose of the aircraft downward). This condition will continue until the aircraft enters a new regime of balance at $\delta_e = \delta_{e\,max}$ (Point 2). In the process of entering this regime, the angle of attack of the aircraft exceeded the critical value and the aircraft stalled.

The influence of a number of factors (geometric or aerodynamic asymmetry of the aircraft, gyroscopic moment of the engine rotor, angles of the control surfaces or ailerons, effect of wing turbulence, etc.) means that the airflow over the aircraft in such regimes is usually asymmetric. Asymmetry of airflow over the aircraft (with respect to the plane of symmetry) leads to the appearance of asymmetric areas of disrupted flow. All of this produces an initial angular velocity of roll $\omega_x \neq 0$. Rolling of the aircraft causes redistribution of the angles of attack on the right and left wings (see Fig. 4.1). In particular, the difference between the angles /127
of attack means that the tangential aerodynamic forces on the right and left wings will be different in value. Under the influence of inertial interaction of the motions, there arises an inertial moment of yaw. All of this leads to the appearance of an angular velocity of yaw $\omega_y \neq 0$. Hence, aerodynamic and inertial moments arise, as well as angular accelerations and angular velocities of rotation of the aircraft relative to all three axes.

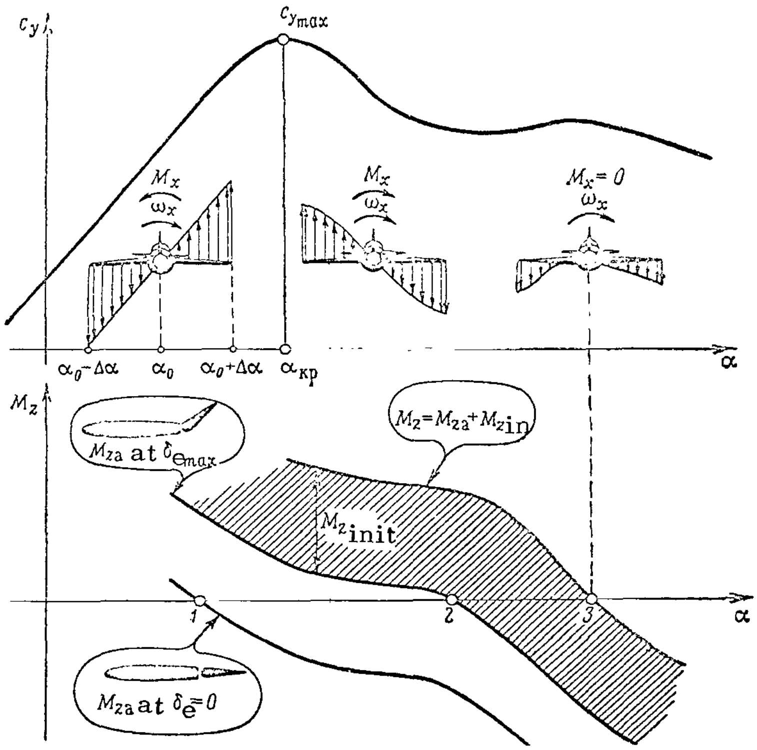

Fig. 4.16. Diagram Illustrating the Development of Motion of an Aircraft From the Original Flight Regime at Subcritical Angles of Attack Until Occurrence of Spin.

The autorotation of the aircraft which arises leads to a significant increase in the initial angular velocity of roll. The inertial moment of pitch $M_{z\ in}$, which appears as the result of an interaction between the longitudinal and lateral motions of the aircraft as it turns, disturbs the balance of moments in the aircraft and shifts it in the direction of larger angles of attack (see Fig. 4.16). In the course of time, with the aircraft moving at supercritical angles of attack, the aerodynamic moments of roll, yaw, and pitch can balance the corresponding inertial moments, while the mass forces (weight, inertial forces) are counterbalanced by the aerodynamic forces. This produces a stable spin regime, i.e., a regime in which all the characteristics of the aircraft's motion remain constant with time. /128

SIMPLIFIED PICTURE OF THE MOTION OF AN AIRCRAFT IN A SPIN

A clear picture of the forces and moments acting on an aircraft in a spin, can be obtained by examining the forces and moments determining the forward motion of the aircraft, i.e., acting in its plane of symmetry. For the sake of simplicity, we will assume that the resultant vector of angular velocity of rotation of the aircraft $\vec{\Omega}$ is also located in its plane of symmetry, while the angular acceleration of pitch at a given moment in time is equal to zero ($\varepsilon_z = 0$). As we did earlier (see Chapter I), we will think of the combination of all the elementary masses of the aircraft m, for the sake of convenience, as two concentrated masses m_1 and m_2 ($m = m_1 + m_2$); located in the nose and tail sections of the fuselage.

As we see from Fig. 4.17, in the case of an airplane in a spin with the engine inoperative, there are four forces and two moments (in the general case) which act in its plane of symmetry. These forces are the weight of the aircraft $\vec{G}$, the centrifugal force of inertia $\vec{F} = \vec{F}_1 + \vec{F}_2$, the tangential force of inertia $\vec{J} = m(d\vec{V}/dt)$, and the resultant aerodynamic force $\vec{R} = \vec{G}_R Sq$. The moments are the aerodynamic moment of pitch $M_{z\ a}$, governing diving, and the inertial moment of pitch $M_{z\ in}$, governing pitching. It is clear from the figure that the force $\vec{R}$ is balanced by the resultant of the three forces $\vec{G}$, $\vec{F}$, and $\vec{J}$. The aerodynamic diving moment is balanced by the pitching moment.

Actually, the motion of the aircraft in a spin is more complex than the simplified scheme presented above. In the general case, all of the characteristics of the motion of an aircraft in a spin are variable with time, and the linear and angular accelerations are not equal to zero. This is due to the very complex inertial-aerodynamic and gyroscopic interaction of the longitudinal and lateral motions of the aircraft in such a regime. Strictly speaking, the above mentioned regime of stable spin can only exist during tests of dynamically similar aircraft models in spin within a wind /129 tunnel (spin in a medium of constant density). A stable regime of spin cannot exist under natural conditions. This is explained as follows:

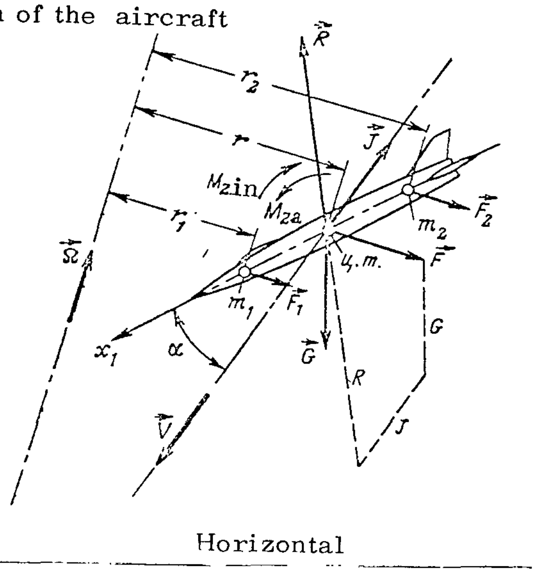

Fig. 4.17. Forces and Moments Acting on An Aircraft in a Spin, in the Longitudinal Plane.

The motion of an aircraft in a spin is related to a change in the flight altitude. A drop in altitude is accompanied by a rise in air density. To preserve a stable regime of spin, it is necessary in particular to preserve constant values for the aerodynamic forces and moments. At fixed angles of attact and sideslip, this means that the value of the reference pressure $q = \frac{\rho n V^2}{}$ must remain constant (if we disregard the effects of compressibility and viscosity, i.e., the Mach and Reynolds numbers for the aerodynamic coefficients). However, the change in the density of the air means that corresponding changes must be made in the actual flight speed in order to keep the velocity head constant. A linear acceleration dV/dt develops. But this is already incompatible with the condition of the existence of a stable regime. Under natural conditions, therefore, the spin regime can only be conditionally stable, i.e., quasistationary. Hence, in the following we will understand a quasistationary regime to be a regime of spin with practically constant (without taking variations into account) average values of the angular velocities, load factors, and instrument flight speed of the aircraft.

4.2. Classification of Spins

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There are a number of different kinds of spin. A given aircraft can be subject to highly diverse spin regimes depending on various factors (flight altitude, alignment, etc.). For rapid and correct orientation in all of these complex and dangerous regimes, and for facilitating the study of spin, as well as evaluating the results of flight tests of modern aircraft in spins, it is useful to divide all of these regimes into several varieties, each of which combines closely related varieties of spin on the basis of their causes. It is useful to use as the basis of such a classification of spins, the following principle: combination of spins by type must be done according to the manner in which they must be escaped. In other words, a given type of spin must include all the regimes which must be escaped by using the same method of pilotage (more detail on this is given in Chapter V).

Depending on the position of the aircraft relative to the ground all spins can be divided into two categories: normal and inverted (Fig. 4.18). A normal spin is one in which the pilot is in a "head up" position (in this respect, such a regime is similar to normal operational flight regimes), i.e., as we have already mentioned above, this is a spin which takes place at positive supercritical angles of attack of the central sections of the wings. An inverted spin is one in which the pilot is in a "head down" position (the aircraft is spinning "on its back", i.e., is in inverted flight), and the aircraft is spinning at negative supercritical angles of attack of the central sections of the wings. Diagrams showing the difference in the position of the aircraft relative to the ground in normal and inverted flight will be found in Figs. 4.19 and 4.20. For increased clarity of the basic features of the regimes, all forms of normal and inverted spin in modern aircraft of conventional design are divided into two basically different groups: stable and unstable spins (see Fig. 4.18).

(a) UNSTABLE SPINS

Unstable spins are those during which the rotation of the aircraft relative to its normal and longitudinal axes periodically changes direction, or stops. Unstable spins are characterized by highly non-uniform rotation with large-amplitude variations of the motion parameters of the aircraft and periodic cessation of the latter. The nose of the aircraft can spontaneously rise above the horizontal position from time to time, or drop to the vertical, and the absolute values of the roll angle can occasionally exceed 90°. In such regimes, there is usually a tendency toward a spontaneous transition of the aircraft from spin in one direction to spin in the other direction, or from normal to inverted flight and vice-versa.

Unstable normal spins are of three types:

(1) Spins occurring in the form of pulses, with periodic increases and decreases of the oscillation of the aircraft:

(2) Spins occurring in the form of a leaf drop, along a spiral trajectory;

(3) Spins, in the course of which the oscillations of the aircraft increase.

Examples of these spins are shown in Figs. 4.21, 4.22, and 4.23. These figures show strip charts from flight recorders, which were obtained during flight tests of supersonic aircraft in spins. In left-hand normal spin involving wobble (see Fig. 4.21), the amplitude of the oscillations of the angular velocities of roll and yaw reach $\Delta\omega_x \approx 4$ rad/sec and $\Delta\omega_y \approx 1.4$ rad/sec at different moments in time. The wobble cycle is about 15 sec. Between these

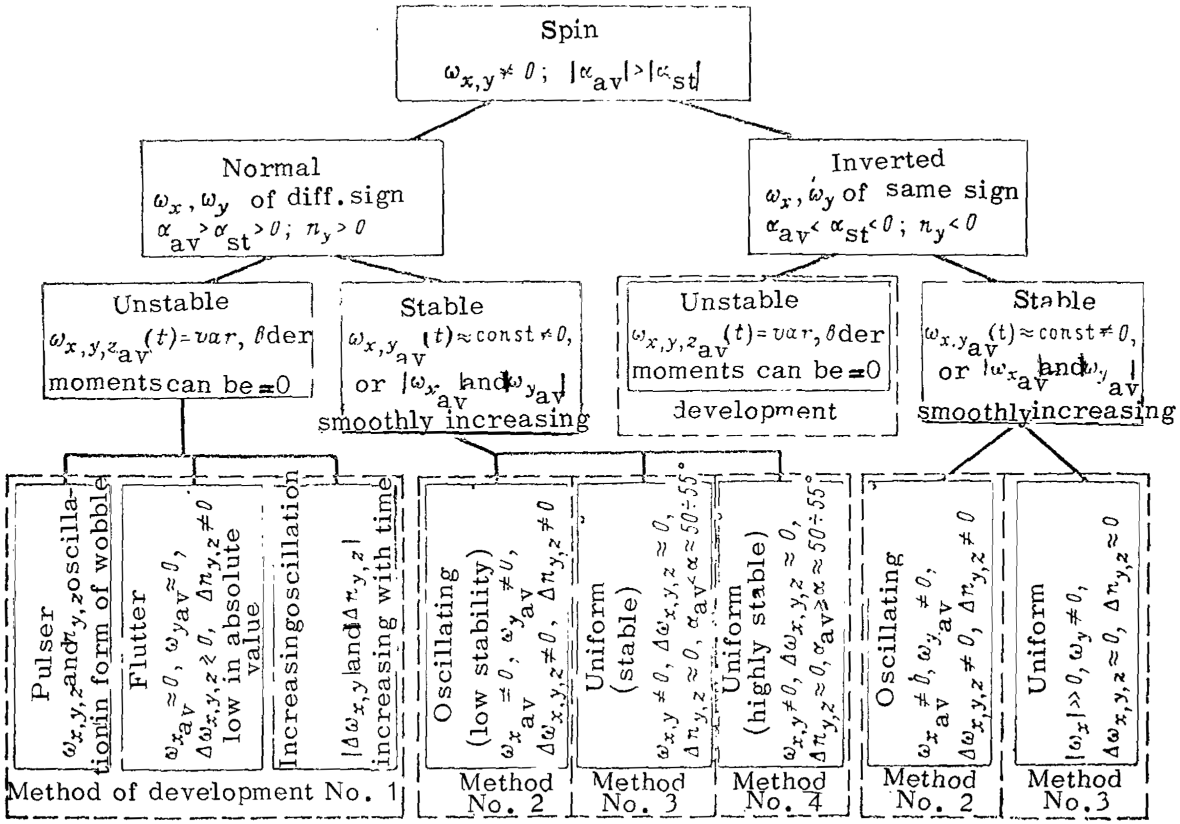

Fig. 4.18. Schematic Classification of Spins in Modern Aircraft.

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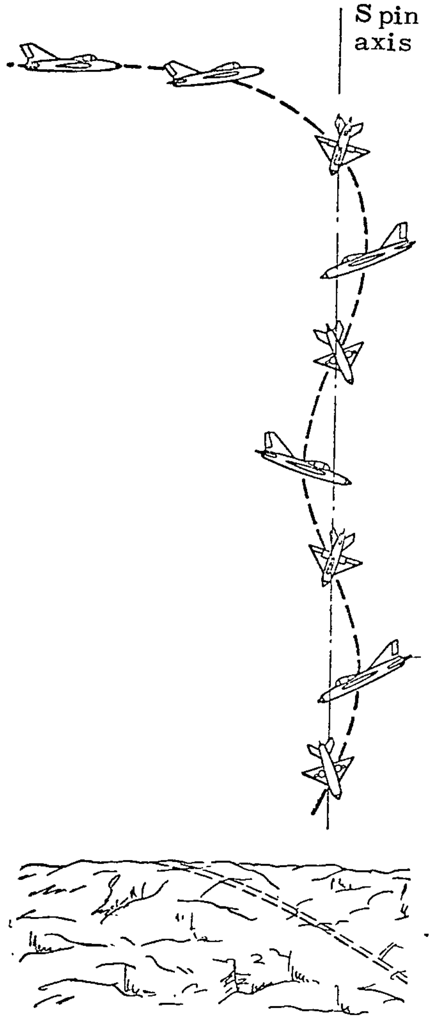

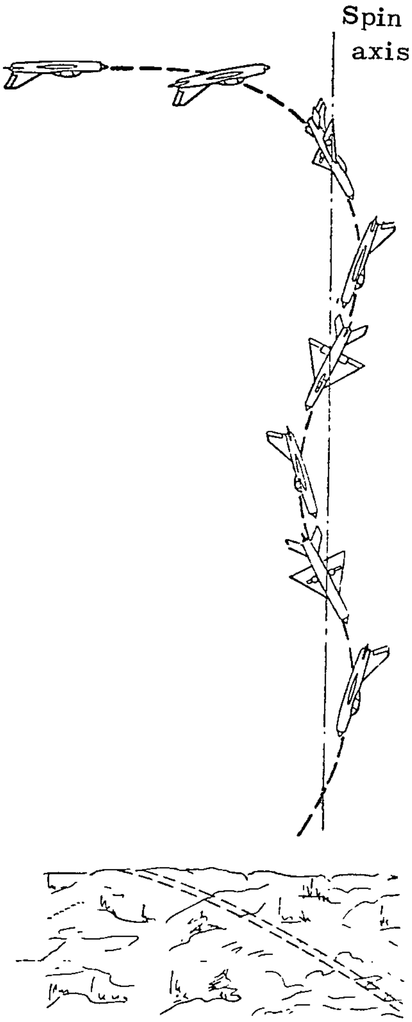

Fig. 4.19 Fig. 4.20

Fig. 4.19. Schematic Representation of the Nature of the Motion of an Aircraft in a Normal Spin. /13

Fig. 4.20. Schematic Representation of the Nature of the Motion of an Aircraft in an Inverted Spin.

cycles, the aircraft is seen to cease rotating relative to its vertical axis ($\omega_y = 0$). Within the limits of such a cycle, the oscillations (or rather, autooscillations) of the aircraft occur with an average frequency on the order of 2.5 sec. /13

The spin regime represented in Fig. 4.22, resembling the fall of a leaf along a spiral trajectory, is accompanied by periodic changes in the rotation direction and abrupt changes in the aircraft's position in space. Thus, for example, during the first two seconds after stalling ($t \approx 8$ to 10 sec), the nose of the aircraft moves to the left, while the aircraft rolls over on its left wing. After a short time ($t \approx 10$ to 11 sec), the nose of the aircraft shifts to the right, then to the left again ($t \approx 11$ to 13 sec), back to the right ($t \approx 13$ to 16 sec, with the angular velocity of roll changing its sign), etc. In other words, this regime involves not only a periodic variation of the magnitude, but also of the signs of the angular velocities of roll and yaw. The aircraft flies as if it were rolling from one wing to the other, with alternate turns of the nose: now to the right, now to the left (like a falling leaf), while its center of gravity moves along a spiral trajectory.

The regime of left-hand normal spin, during which increasing oscillations arise (autooscillations) in the aircraft, is shown in Fig. 4.23. It is clear from the graph that a definite increase in the oscillations of normal force began approximately 10 sec after the aircraft stalled. At the end of this time, the average value of the normal force began to increase monotonically, and its oscillations relative to its average value increased rapidly. For example, 30 seconds after they began, the amplitude of the oscillations reached $\Delta n_y \approx 5$. During these oscillations, which occurred with a period on the order of two seconds, the aircraft flew now at small subcritical (value of n_y close to zero), now at large supercritical angle of attack.

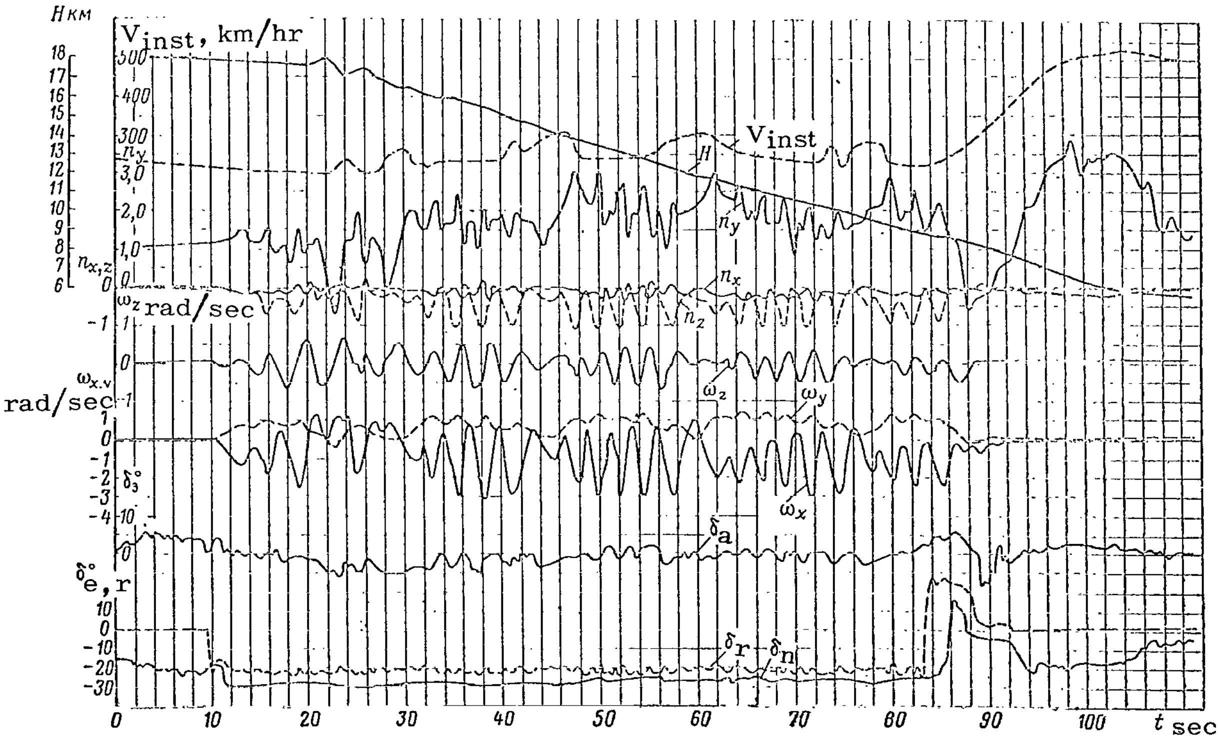

Fig. 4.21. Example of Unstable Left-Hand Normal Spin, Occurring in the form of Pulser. /134

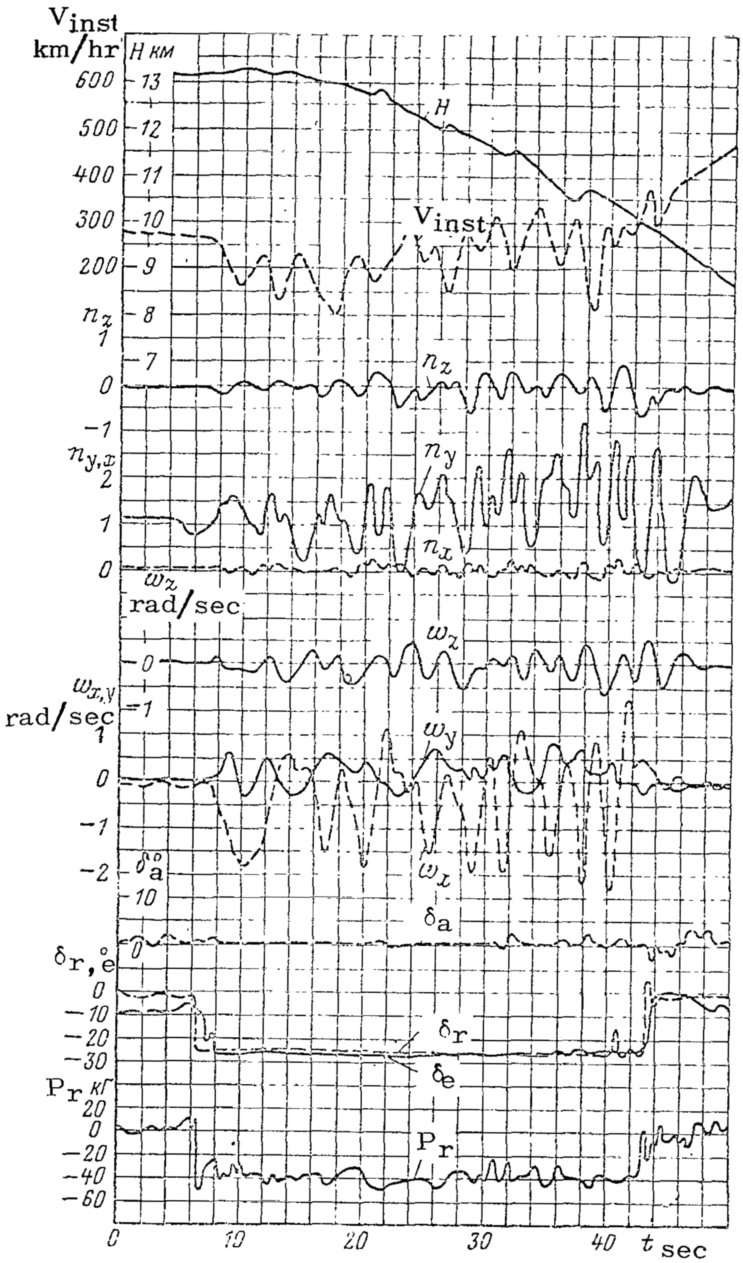

Fig. 4.22. Example of Unstable Normal Spin, Occurring in the Form of a Leaf Drop Along a Spiral Trajectory (Δt $\approx$ 9 to 42 sec.). /135

It should be noted that the values for the instrument speed and flight altitude shown on these graphs, obtained during spin and recorded on the strip charts of the flight recorders, do not represent the actual values of these parameters. This is due to /136

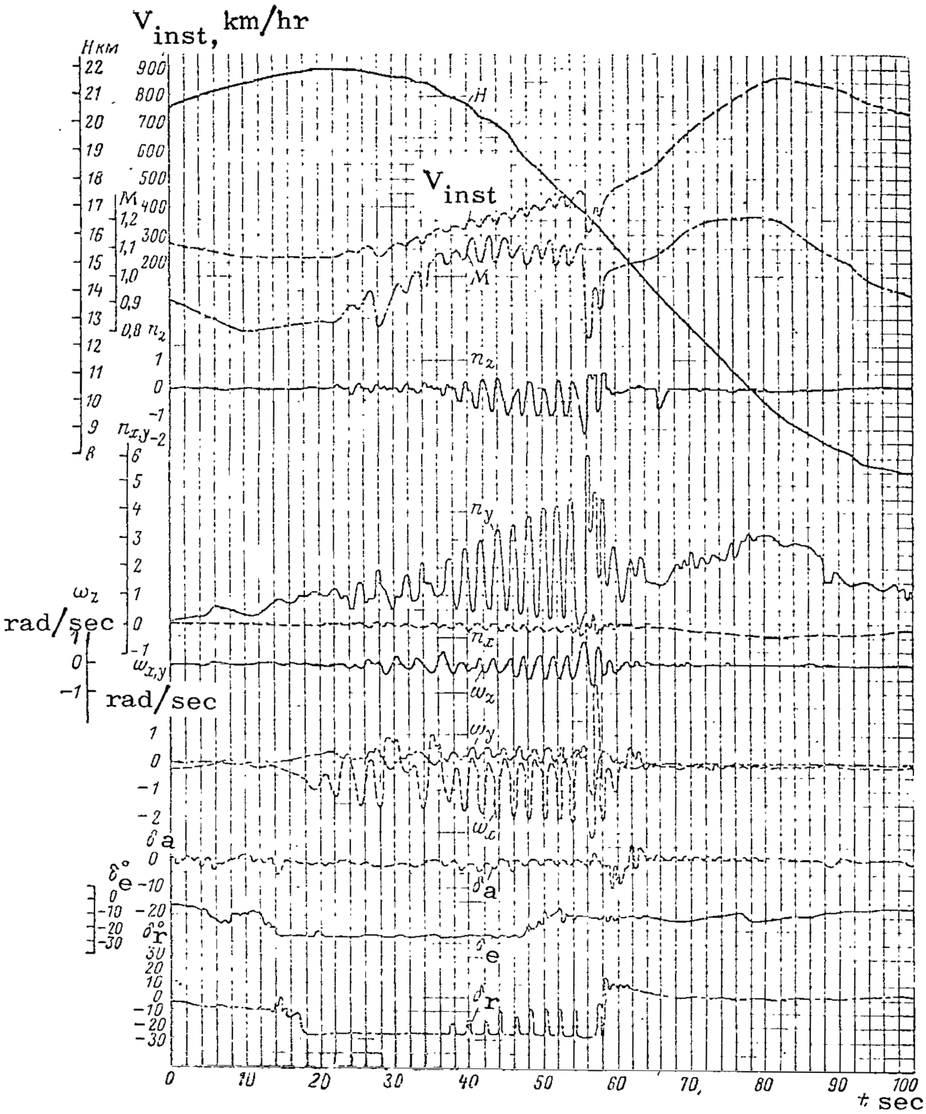

Fig. 4.23. Example of Unstable Left-Hand Normal Spin, Involving Increased Oscillation of the Aircraft ($\Delta t \approx 19$ to 59 sec).

the fact that at large supercritical angles of attack, the readings indicated by the usual airspeed indicator aboard an aircraft (which shows the static air pressure as well as the velocity head of the flow) involve large errors, which cannot be taken into account in practice. In particular, the variations in the instrument speed and flight altitude during spin, which are indicated on the graphs, /137 are mainly the result of errors in the instrument, caused by variations of the angle of attack of the aircraft. The values of V_{inst} and H shown on the graphs serve only to provide an approximate qualitative idea of the nature of the changes in these values in a spin.

In the majority of cases, regimes of unstable spin can exist for a long period of time with the control surfaces deflected during spin, and cease only when the control surfaces are returned to a neutral position. In some cases, however, the aircraft can spontaneously escape from spin even with the control surfaces deflected. This involves an increase in the flight speed during the spin. The *flight regime in which the motion of the aircraft takes place with* successive alternations of entrance into a region of supercritical angles of attack and emergence from it, with a constant increase in the average values of the instrument flight speed and the normal load factor, is called a progressive spiral spin. The increase in the normal load factor during oscillation of the aircraft in such regimes can reach values that are dnagerous from the standpoint of the aircraft's durability. Hence, regimes of this kind must not be allowed to continue for a long period of time: the pilot must attempt to pull the aircraft out of such a regime as soon as possible. A clear example of such a regime is shown in Fig. 4.24, an unstable normal spin involving flutter, with the aircraft falling like a leaf along a spiral trajectory in the form of a progressive spiral spin.

Unstable normal spin usually arises after stall at large (or rarely, average) flight altitudes at the beginning of the regime. With a high degree of longitudinal static stability of the aircraft under force (forward alignment) and incomplete movement of the control stick backward, unstable normal spin can even occur at low altitudes. Unstable inverted spin arises, as a rule, when the aircraft stalls at a great altitude when flying upside down or nearly so (stall at negative angles of attack), and also in the case of spontaneous entrance of the aircraft into unstable normal spin at negative supercritical angles of attack. An example of such spin is shown in Fig. 4.25. Flying experience has shown that regimes of unstable inverted spin are encountered relatively rarely in supersonic aircraft.

(b) STABLE SPINS

Stable spins are those in which the aircraft does not change the direction of its motion, either in yaw (unchanging sign of the angular velocity of yaw, ω_y) nor in roll (unchanging sign of the average value of the angular velocity of roll $\omega_{x_{av}}$) and in which the pilot does not sense any suspension of motion. The latter phenomenon in such regimes is relatively (an sometimes very) intense and stable.

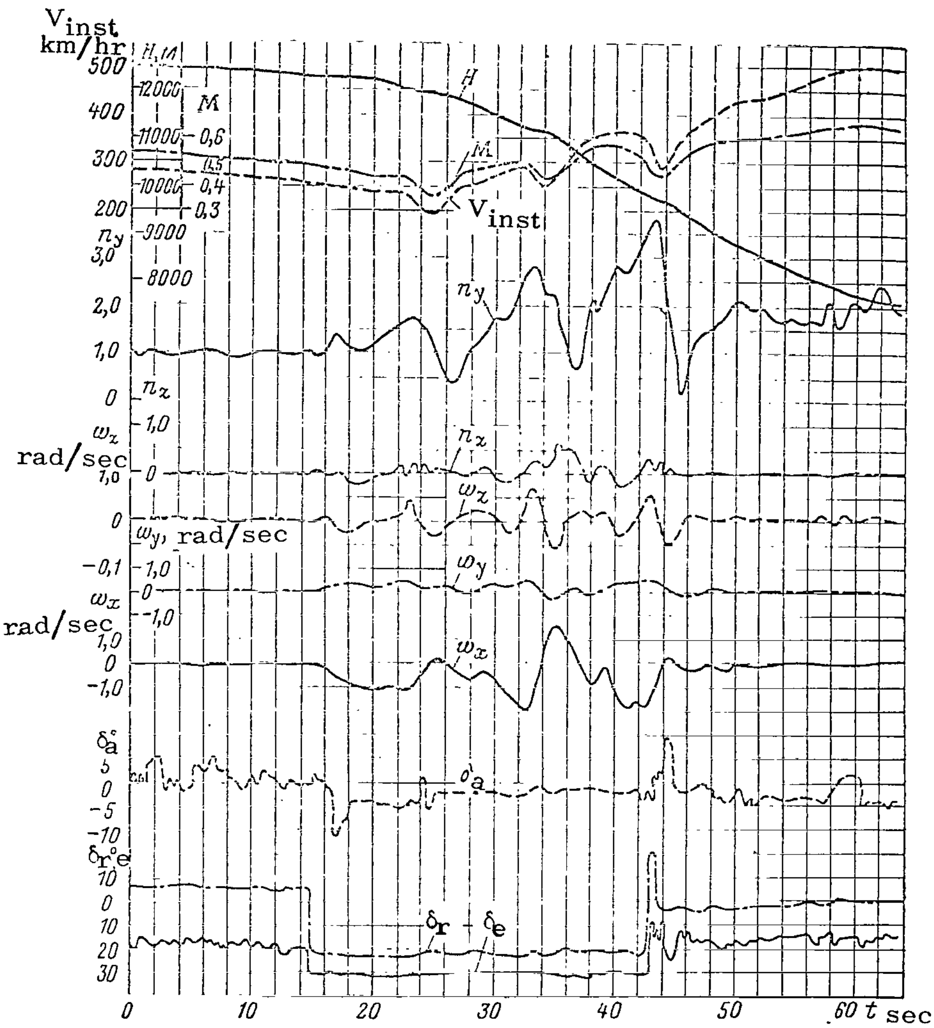

Fig. 4.24. Example of Progressive Spiral Spin ($\Delta t \approx$ 17 to 44 sec). /138

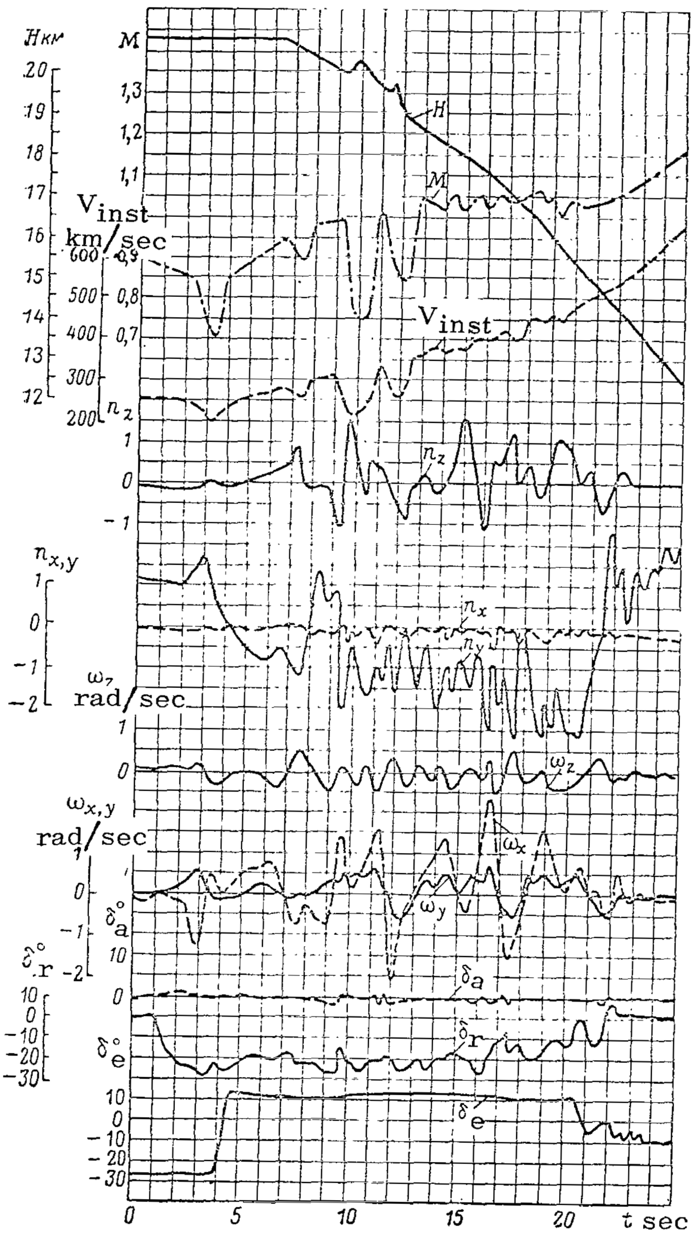

Fig. 4.25. Example of Unstable Inverted Spin ($\Delta t \approx$ 3 to 23 sec). /139

Stable spins can be either oscillating or uniform. Stable oscillating spin (a relatively unstable regime) may be accompanied /140

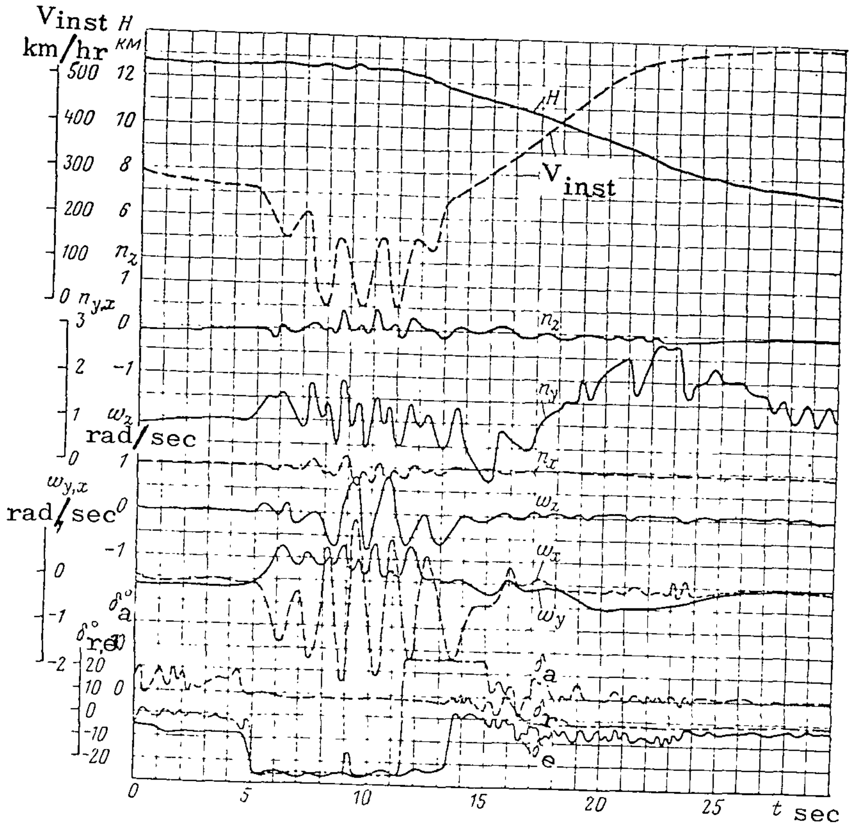

Fig. 4.26. Example of Stable Oscillating Normal Spin ($\Delta t \approx 6$ to 16 sec).

by variations in the angular velocity of roll that are very large in amplitude (Figs. 4.26 and 4.27), but then the amplitudes of the oscillations in the angular velocity of yaw are relatively slight. The amplitudes of the oscillations of the vector component of stress n_x, n_y and n_z in stable spin usually turn out to be much less than in unstable spin.

Stable oscillating normal spins are found as a rule at middle altitudes at the beginning of the regime, or with a relatively small duration of the regime, i.e., in the transitional part of the spin (see section 4.3). The aircraft can be placed in stable, oscillating, inverted spin by tilting the ailerons in a regime of stable oscillating normal spin or by pilotage errors made in attempting to pull the aircraft out of unstable or stable oscillating normal spin. In stable inverted spin, the average absolute values of the angular velocities of roll and yaw are usually relatively

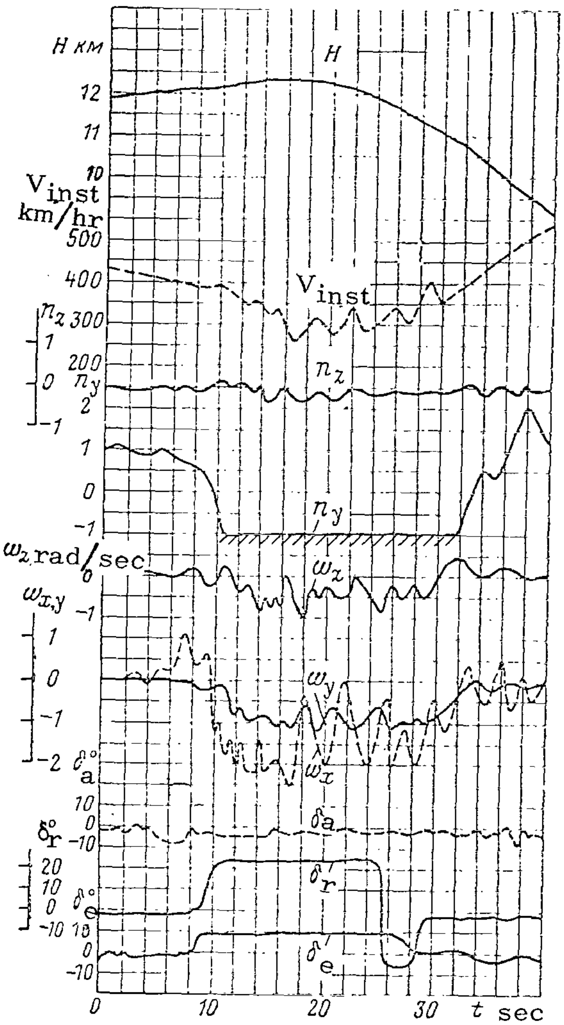

Fig. 4.27. Example of Stable Oscillating Inverted Spin. ($\Delta t \approx 18$ to 30 sec). /141

close to one another. Thus, in particular, in the case of the example shown in Fig. 4.27 ($t \approx 18$ to 30 sec), the average value of the angular velocity of roll amounts to about $\omega_{x_{av}} \approx -1.2$ rad/sec, while that of the angular velocity of yaw is on the order of $\omega_{y_{av}} \approx 0.95$ rad/sec. /142

Stable uniform spins occur at relatively small (sometimes practically zero) oscillation amplitudes of the aircraft and are characterized by an intense and steady rotation of the aircraft, with a stable maintenance of the initial direction of this rotation. Stable uniform normal spins are of two kinds (see Fig. 4.18): with a moderate degree of stability, and highly stable. Stable uniform spin which has a moderate degree of stability, usually takes place at average angles of attack in the α_{av} regime which are less than 50 to 55°; a large number occur at α_{av} on the order of 45° (for the determination of α_{av}, see section 4.3). Highly stable uniform spin occurs, as a rule, at average angles of attack greater than 50 to 55°, usually about 60 to 70° but sometimes more. Such highly stable spin involves very intense rotation of the aircraft (sic).

An example of uniform normal spin of the first type is given in Fig. 4.28 ($\alpha_{av} \approx 45°$), while the second type can be seen in Fig. 4.29 ($t \approx 35$ to 45 sec, $\alpha_{av} \approx 60°$). Stable uniform normal spin usually arises at moderate and low altitudes, especially with long duration of the regime (vertical spin, about which more will be said in the following paragraph). As we have already pointed out, spins of this kind occur in aircraft at high supercritical angles of attack: $\alpha_{av} \gg \alpha_{st}$, so that $\alpha_{av} \geq 45°$ as a rule.

Stable uniform inverted spin usually occurs at very high absolute values of angular velocity or roll (on the order of 3 rad/sec and more) with relatively low oscillation of the aircraft. It usually arises as a result of pilotage errors in pulling the aircraft out of a stable uniform normal stall, and more rarely, immediately after stalling of the aircraft at initial negative angles of attack at moderate and low flight altitudes. Examples of stable uniform inverted spins are given in Figs. 4.30 ($t \approx 29$ to 38 sec) and 4.31 ($t \approx 53$ to 59 sec).

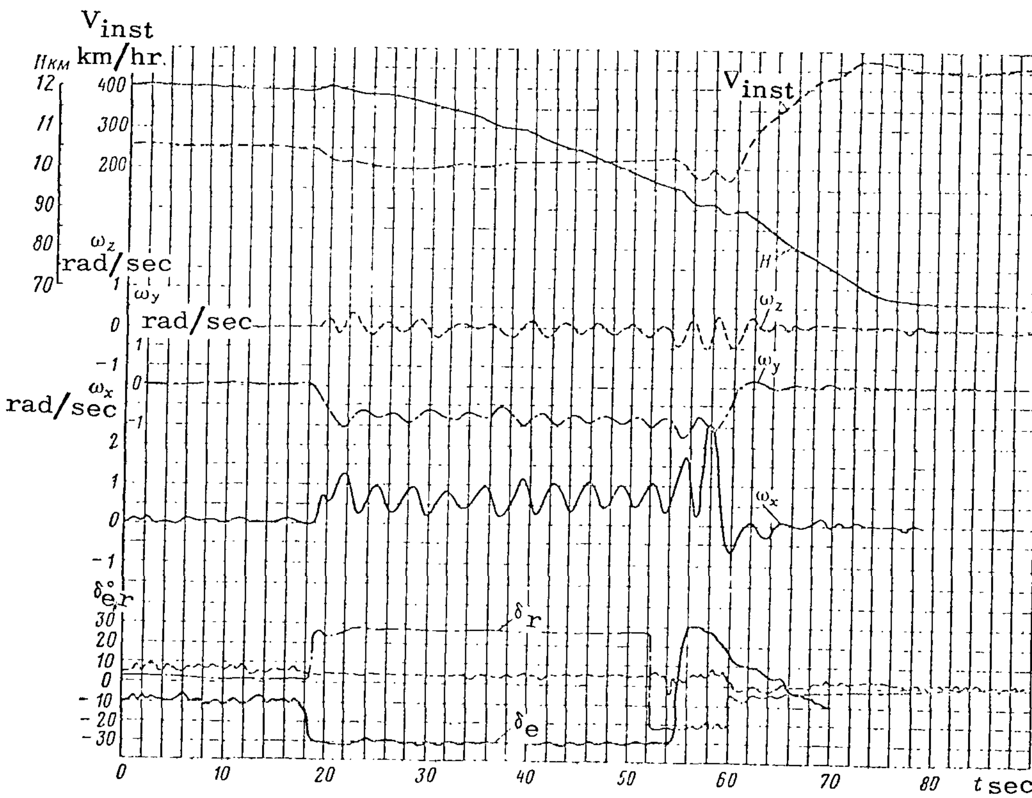

Fig. 4.28. Example of Stable Uniform Normal Spin ($\Delta t \approx 21$ to 61 sec). /143

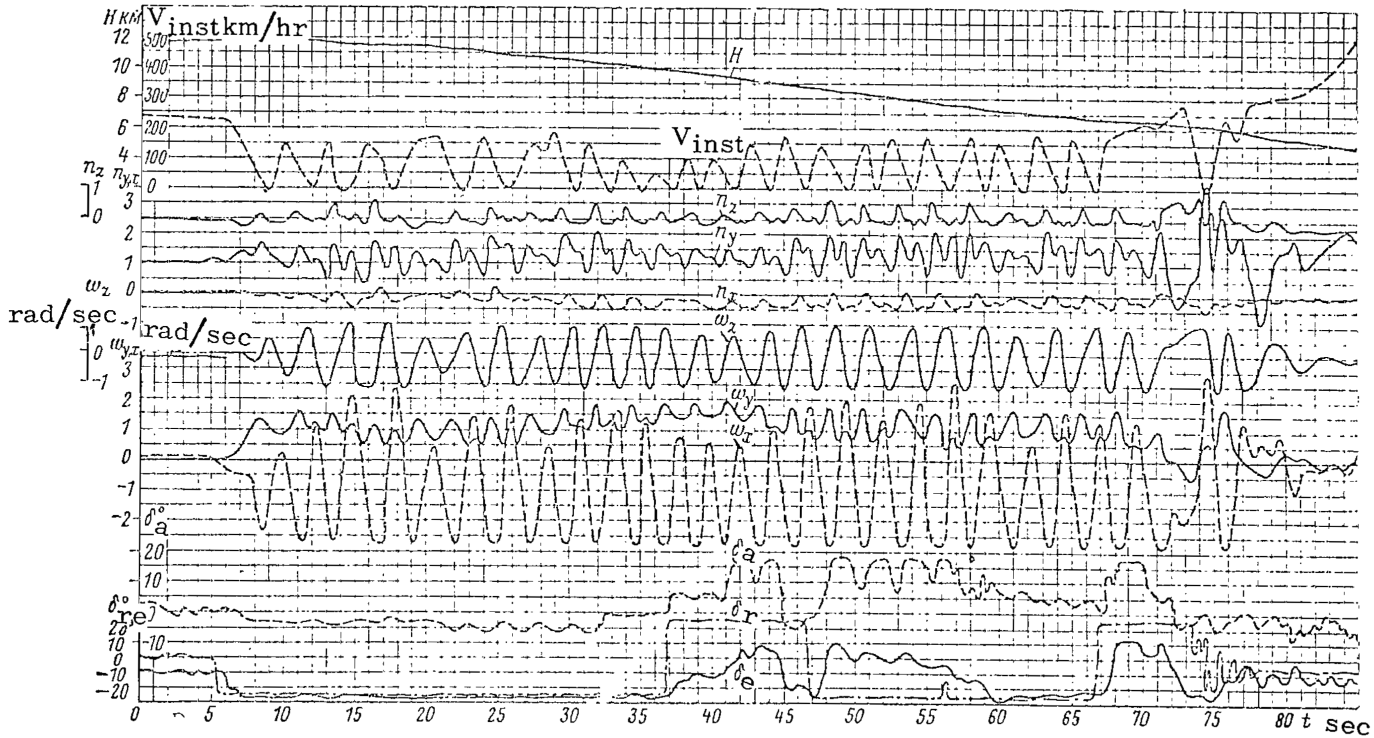

Fig. 4.29. Example of Very Stable Uniform Normal Spin at α_{av} = 60° ($\Delta t \approx$ 35 to 45 sec).

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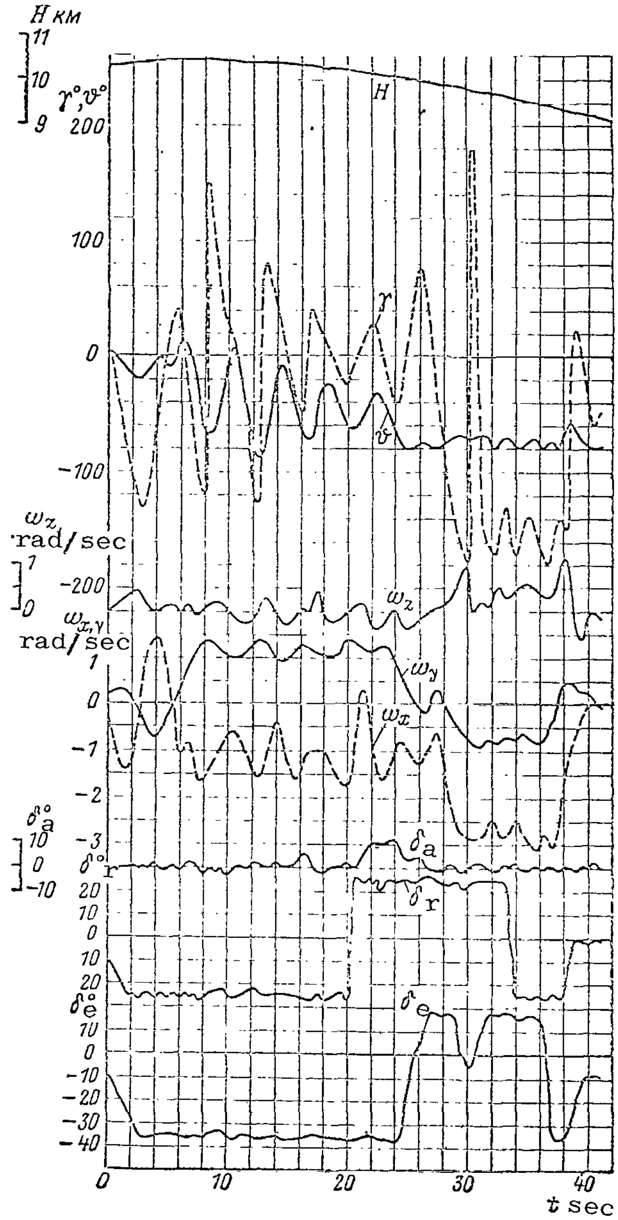

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Fig. 4.30. Example of Stable Uniform Right-Hand Inverted Spin ($\Delta t \approx$ 29 to 38 sec).

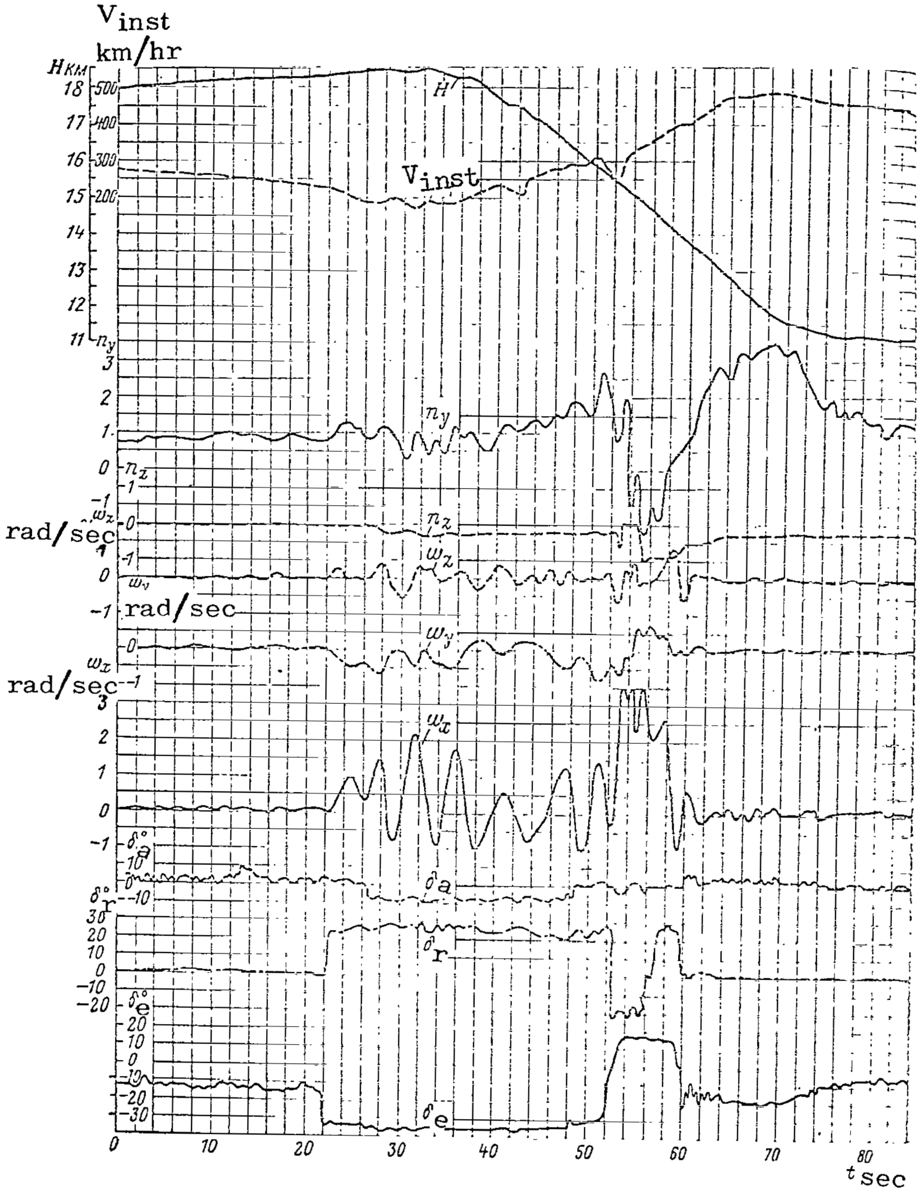

Fig. 4.31. Example of Stable Uniform Left-Hand Inverted Spin ($\Delta t \approx 53$ to 59 sec).

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4.3. Peculiarities of Spin in Supersonic Aircraft /147

The spin regime may be thought of as consisting of two stages (Fig. 4.32): an incipient section, and vertical spin. The transitional section of spin is that part of the regime which lasts from the moment when autorotation arises following stalling of the aircraft, until the moment when the spin axis becomes practically vertical. The spin axis is the axis (midline) of spiral spin, i.e., the spiral which forms the trajectory of the aircraft's center of

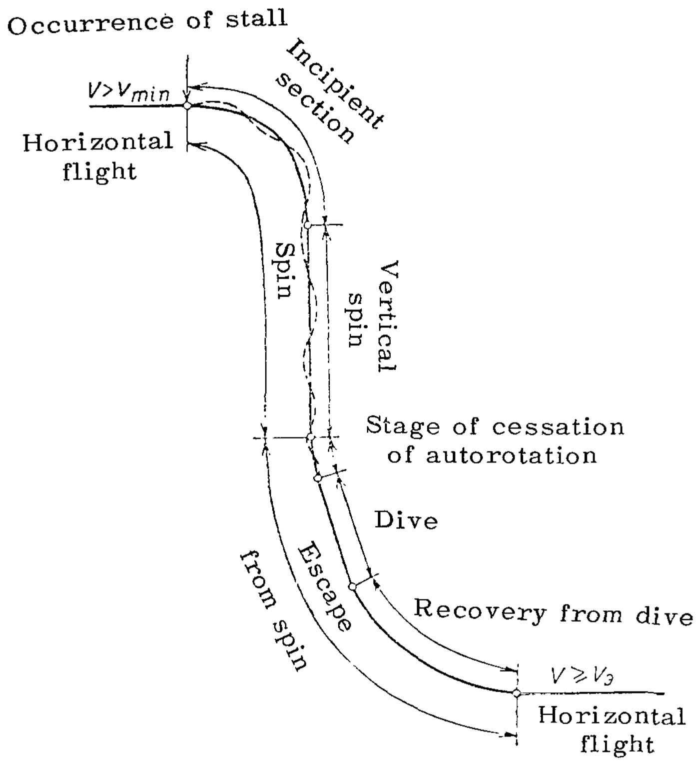

Fig. 4.32. Schematic Representation of the Component Stages in Spin and Escape from It.

gravity. Vertical spin begins at the end of the transitional section and usually lasts until the aircraft begins to escape the spin. /148

Before proceeding to a discussion of the features of each of these stages of spin in detail, let us find the formula for determining the average angle of attack and the sideslip angle of an aircraft in a vertical spin, which are the most important characteristics of the regime.

(a) TRANSITIONAL SECTION OF SPIN

With uniform initial conditions (H_0, n_{y0}, etc.), the transitional section of spin in supersonic aircraft is much more prolonged than in subsonic aircraft, due mainly to an increase in the flying weights and minimum velocities in supersonic aircraft (caused by an increase in the wing loading and a drop in the carrying capacity of the wings). This is explained as follows:

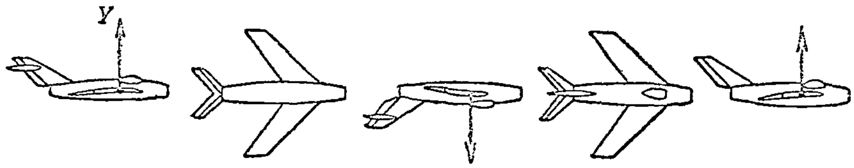

Fig. 4.33. Diagram of Appearance of Autoneutralization of Lift During Rapid Rotation of an Aircraft.

Let us consider schematically the motion of an aircraft after stalling from an initial regime of straight-line horizontal flight. To estimate the qualitative aspect of the phenomenon, let us assume for the sake of approximation that in the presence of autorotation the vector of lift of the aircraft, which is rotating along with it, will be directed upward part of the time and downward part of the time (Fig. 4.33). In fact, if the absolute value of the angular velocity of roll is sufficiently great, and its average value is constant or changes relatively slowly, the lift of the aircraft will appear to neutralize itself, and the net effect of the action of the lift in such a case will be practically zero, as if the lift were completely nonexistent. This effect is called autoneutralization of lift.

Autoneutralization of lift, i.e., with its net effect equal to zero, naturally does not exclude the influence of lift on the trajectory of the aircraft at any given moment in time. In fact, then, the trajectory of an aircraft in the transitional section of spin is a helix, whose axis can be considered (in approximation) as the trajectory of a freely falling body. When sufficiently large angular velocities of roll are obtained, both of these trajectories will come closer together, to the point where it becomes possible (for the sake of approximation) to consider the trajectory /149
of the aircraft's center of gravity in the transitional section of the spin as the trajectory of a freely falling body of the same weight and the same initial velocity. Hence, we can assume for the sake of approximation that the trajectory of the motion of an aircraft, after stalling initially, resembles a parabola, then gradually approaches a vertical straight line. The greater the initial flight speed and weight of the aircraft (the greater its reserve of kinetic energy), the longer this transition will last and the longer will be the flying distance L and the loss of altitude ΔH in the transitional section of the spin (up to the point when the trajectory of the center of gravity of the aircraft approaches vertical).

Fig. 4.34 shows the calculated trajectories of an aircraft after stalling from various speeds, obtained under the condition $c_y = 1.25 = \text{const}$, $c_x = 1.25 = \text{const}$ and $G/S = 250\ \text{kg/m}^2$ for various values of the angular velocity of roll (Curves 1,2, and 4). Here, however, for the sake of comparison we have plotted the parameters (obtained under these conditions) of the trajectory of the aircraft's center of gravity, viewed as a freely falling body (in the absence of lift: $c_y = 0$, Curve 3). Comparison of these curves shows that at an angular velocity of roll $\omega_x = 1.5$ rad/sec, the trajectory of the aircraft's center of gravity after stalling is already very close to the trajectory of a freely falling body.

As the altitude for the beginning of stall increases, the transitional section of spin becomes more sloped and its length increases, which is caused by an increase in the initial true flight speed at which the stalling of the aircraft occurs: at a constant reference pressure ($q = \text{const}$), an increase in altitude due to a drop in air density means that the true flight speed will increase. Increase of the true flight speed in turn evokes an increase in the Mach number. A drop in $c_{y\,\mathrm{st}}$ under the influence of an increase in the Mach number at high altitudes will cause an additional increase in the flight speed at which stalling occurs.

In view of the relatively long duration of the transitional section of spin, especially after stalling at high altitudes, the pilot of a supersonic aircraft that has gone into a spin must deal mainly with this transitional section, and not with the regime of vertical spin. In the majority of cases, the pilot begins to pull his aircraft out of the spin long before the vertical section of it has begun. Therefore, the importance of the details of guiding and piloting a supersonic aircraft in the transitional section of the spin is considerably greater.

The motion of a supersonic aircraft in the transitional section of a spin occurs usually very irregularly, with large oscillations, sometimes with interruptions and even changes in the rotation direction. The results of flight and ground tests indicate that this is caused by the effect of a nonlinear shape of the aerodynamic characteristics for the angles of attack and sideslip, the Mach and Reynolds numbers, etc., as well as by the action of the gyro- /150
scopic moment of the engine and the effect of non-coincidence of the axis of rotation of the aircraft with the direction of the flight speed vector.

Non-coincidence of the axis of the aircraft's rotation with the direction of the flight speed vector produces a change in the angles of attack and glide during rotation of the aircraft (see Chapter I). A significant change in the flight speed in the transitional section leads to a corresponding change in the Mach and Reynolds numbers. In the presence of nonlinear aerodynamic characteristics, the changes in these parameters lead to a noticeable but mainly nonproportional, non-linear change in the aerodynamic

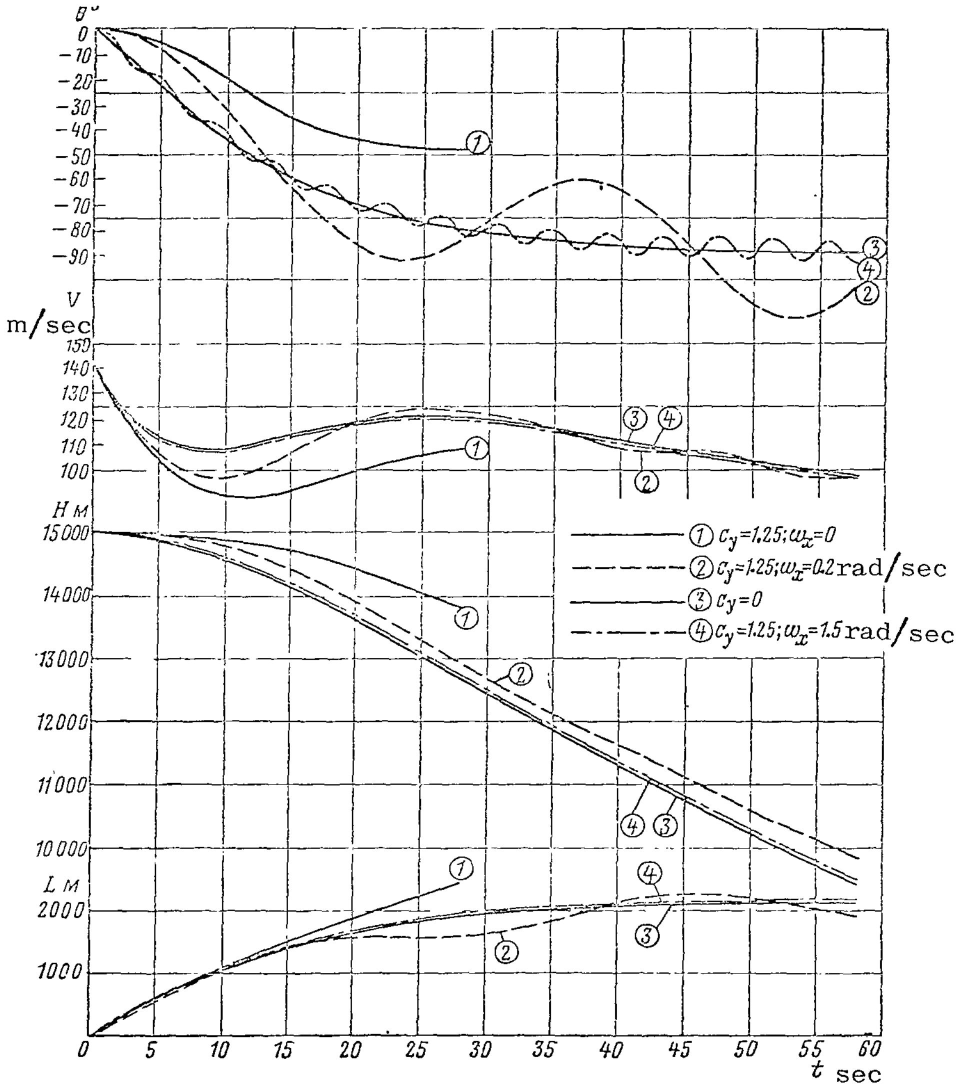

Fig. 4.34. Effect of Angular Velocity of Rotation of an Aircraft on the Trajectory of its Center of Gravity after Stalling.

moments and forces. These changes in the aerodynamic moments and forces, as will be pointed out below, can occur very abruptly and unevenly. Such changes can also lead to the occurrence of nonuniform motion of the aircraft, with considerable oscillations. The changes which then arise in the angular velocities of yaw and roll cause changes in the corresponding components of the gyroscopic moment, which are in agreement with the appearance of still greater irregularities in the motion of the aircraft.

The end of the transitional section and the beginning of vertical spin usually occur in connection with the change in the regime characteristics, well-known to pilots. At the end of the transitional section (this roughly corresponds to the beginning of stable or rather quasistable descent of the aircraft), the rotation of the aircraft becomes more regular, intense, and uniform.

The nonlinear character of the curve of the aerodynamic coefficients and their derivatives for the angles of attack and sideslip and the Reynolds and Mach numbers in supersonic aircraft can be illustrated by the following examples.

Figures 4.35 and 4.36 show examples of the change in the coefficients of aerodynamic forces and moments of aircraft with sweptback wings, as a function of the angles of attack and sideslip The change in the derivatives of the coefficients of the aerodynamic moments as a function of the Mach number was discussed in Chapter I, but the change in the coefficients of the aerodynamic forces as a function of this parameter is illustrated by the polars plotted in Fig. 4.37. It is clear from the graph in Fig. 4.37 in particular that a typical feature of the changes in the polars with transition from subsonic to high supersonic flight speeds is an increase in the coefficient of drag with $c_y = 0$, caused by an increase in the profile-wave and harmful drag of the nonsupporting parts of the aircraft, and a decrease in the slope of the polars relative to the abscissa, caused by an increase in the inductive reactance with a rise in the Mach number.

Examples of the curves of aerodynamic coefficients as functions of the Reynolds number are shown in Figs. 4.38 and 4.39. It is clear from all of these graphs that the curves of these parameters are essentially nonlinear. Such a nonlinearity of the aerodynamic characteristics can lead to important changes in the parameters of motion of an aircraft in a spin.

INFLUENCE OF THE POSITION OF THE ROTATIONAL AXIS /152

Let us examine the influence of the position of the rotational axis of the aircraft on the characteristics of the transitional section of spin. It was pointed out in Chapter I that the rotation of the aircraft (in the general case) occurs relative to an axis which does not coincide with the direction of its longitudinal axis or with the direction of the flight speed vector. Changes in the angles of attack and sideslip which arise as the aircraft rotates, and the axis of rotation of the aircraft will be very close to, or even practically coincident with, its longitudinal axis, i.e., with the initial position of the rotational axis. In the latter case, the aircraft can be viewed as a gyroscope, tending to keep its rotational axis fixed in space.

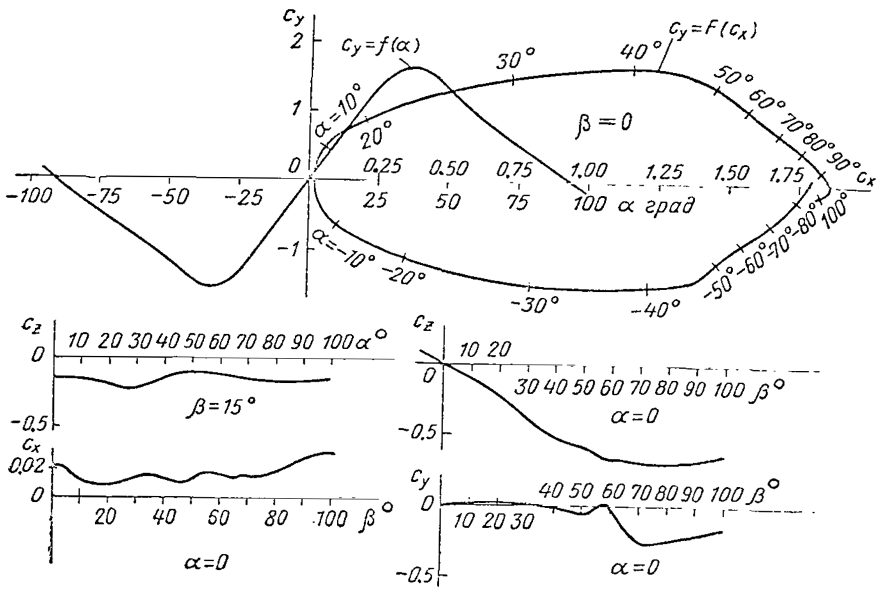

Fig. 4.35. Example of the Change in the Coefficients of the Aerodynamic Forces of an Aircraft with Sweptback Wings, as a Function of the Angles of Attack and Sideslip (M = 0.1; Re = 500,000).

Hence, the larger the ratios $|M_{x\,a}|:|M_{y\,a}|$ and $|M_{x\,a}|:|M_{z\,a}|$, and therefore the ratios of the angular velocities $|\omega_x|:|\omega_y|$ and $|\omega_x|:|\omega_z|$, the closer the rotational axis of the aircraft will be to its longitudinal axis. Due to the increase of the mass differential in the direction of the longitudinal axis in supersonic aircraft, roll will develop more rapidly than yaw and pitch, leading to an increase in these ratios of the angular velocities in the transitional section of spin.

In view of the fact that the rotational axis of supersonic aircraft is located closer to its longitudinal axis than in subsonic aircraft, while the critical angles of attack are much greater, the kinematic changes in the angles of attack and sideslip during rotation of a supersonic aircraft after stalling increase. This is explained by the following example:

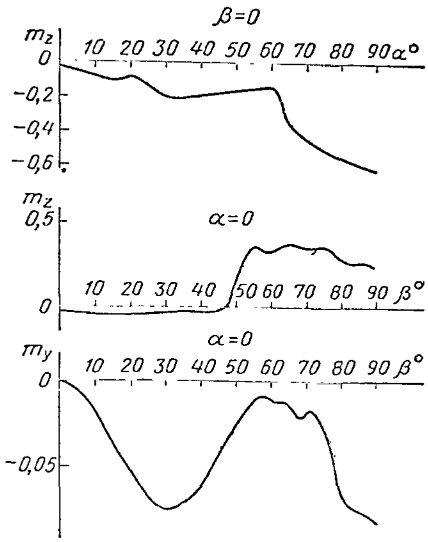

Fig. 4.36. Example of the Change in the Coefficients of the Aerodynamic Moments of an Aircraft with Swept-back Wings as a Function of the Angles of Attack and Sideslip (M = 0.1; Re = 500,000).

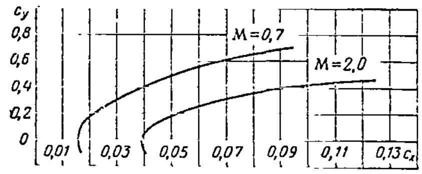

Fig. 4.37. Polars of an Aircraft with Sweptback Wings at Low Subsonic and High Supersonic Mach Numbers (β = 0; Re = 500,000). /153

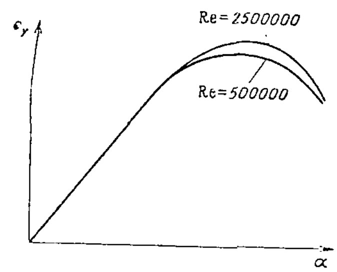

Fig. 4.38. Effect of the Reynolds Number on the Curve $c_y = f(\alpha)$ in an Aircraft with Sweptback Wings (M = 0.1; β = 0).

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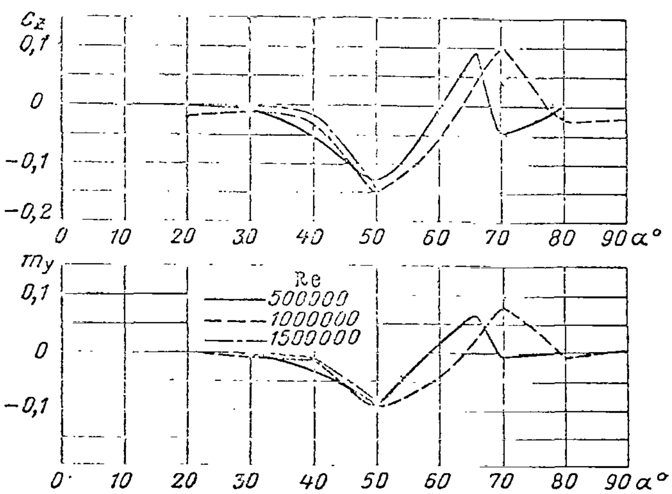

Fig. 4.39. Effect of Reynolds Number on the Curves $c_z = \phi(\alpha)$ and $m_y = F(\alpha)$ in an Aircraft with Sweptback Wings (M = 0.1; β = 0).

The stall angles in supersonic aircraft $\alpha_{st\ super}$ are much greater than the stall angles in subsonic aircraft, $\alpha_{st\ sub}$. Since the rotational axis $O\zeta$ in supersonic aircraft (with all other conditions being equal) is located closer to the longitudinal axis ox_1 than is the case for subsonic aircraft, the angles ψ between these axes in supersonic aircraft are smaller: $\psi_{super} < \psi_{sub}$ (Fig. 4.40). Following a stall, therefore, the angle between the rotational axis of the aircraft and the direction of the flight speed vector in supersonic aircraft (ϕ_{super}) is much greater than in subsonic aircraft (ϕ_{sub}): $\phi_{super} \gg \phi_{sub}$. This means, however, that the changes in the angles of attack and sideslip during rotation of a supersonic aircraft will be greater than in rotation of subsonic aircraft.

Fig. 4.41 is a schematic representation of the changes in the angles of attack which occur when the aircraft turns through 360° relative to the axis of rotation $O\zeta$. We see from Figs. 4.40 and 4.41 that the greater the values α_{st} and $\phi-\psi$, the greater will be the changes in the angle of attack during the rotation of the aircraft.

In view of the fact that in supersonic aircraft the rotational axis $O\zeta$ is close to the longitudinal axis ox_1, when we consider the kinematic changes in the angles of attack and sideslip, in the first approximation it is possible to consider the rotation of the aircraft as taking place relative to its longitudinal axis. The assumption of such a simplification permits us to draw the qualitative picture of the phenomenon more smoothly. Figure 4.42 shows the progress of kinematic changes in the angles of attack and sideslip during rotation of an aircraft through 360° relative to its longitudinal axis ox_1. This diagram, in particular, shows clearly why in the case of supersonic aircraft in the transitional section of a spin, the extreme values of the sideslip angle β, as a rule, are displaced with respect to the extreme values of the angle of attack α by about 1/4 of a revolution. /155

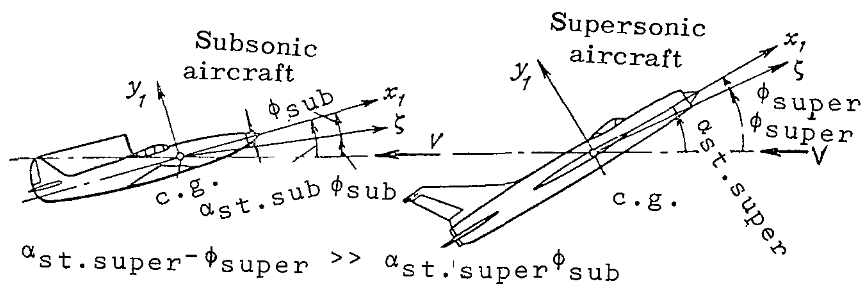

Fig. 4.40. Diagram Showing the Reasons for an Increase in the Kinematic Changes in the Angles of Attack and Sideslip in a Supersonic Aircraft.

Under the influence of the force of gravity (the lift practically neutralizes itself, see Fig. 4.33), the trajectory of motion,

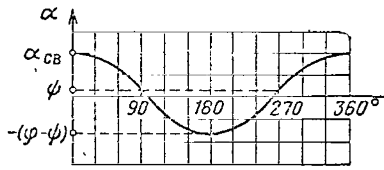

Fig. 4.41. Diagram of Kinematic Changes in the Angle of Attack with Rotation of the Aircraft Relative to the Axis 0ζ.

i.e., the flight speed vector of the aircraft in the transitional section of the spin, moves downward. Supersonic aircraft, much more than subsonic aircraft, have a characteristic tendency to retain the initial direction of the rotational axis. Like a gyroscope, the "spinning" aircraft attempts to keep its axis of rotation fixed in space. The large ratio J_z/J_x makes it much easier for the aircraft to spin on its longitudinal axis ox_1 than on its transverse axis oz_1.

However, due to the large angle of attack of the empennage (a large angle between the axis ox_1 and the flight speed vector $\vec{V}$), a relatively greater aerodynamic force acts on the latter, tending to push the nose down. The nose of the aircraft drops slowly under /156

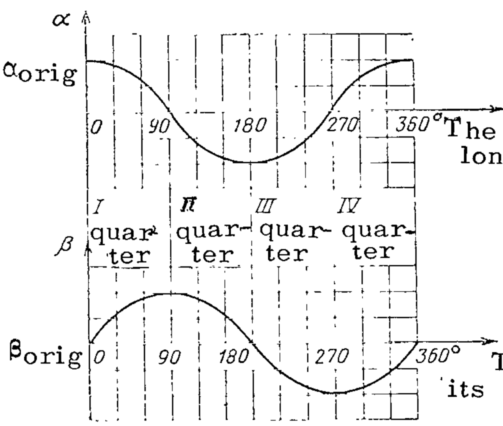

Fig. 4.42. Diagram of Kinematic Changes in the Angles of Attack and Sideslip with Rotation of the Aircraft Relative to its Longitudinal Axis.

the influence of these forces, but the rotational axis of the aircraft follows the changing direction of the speed vector (but with some lag) (Fig. 4.43). Ultimately (at average altitudes, usually in 15 to 20 sec, but occasionally longer), the speed vector and then the rotational axis of the aircraft assume a vertical position or one very nearly so, i.e., their directions practically coincide, and vertical spin begins.

In connection with the increase in drag of the aircraft c_x at supercritical angles of attack after stalling, the flight speed begins to drop, thus leading to a reduction of the values of the aerodynamic moments. However, since the resultant aerodynamic moments are proportional to the square of the speed, while the auto-

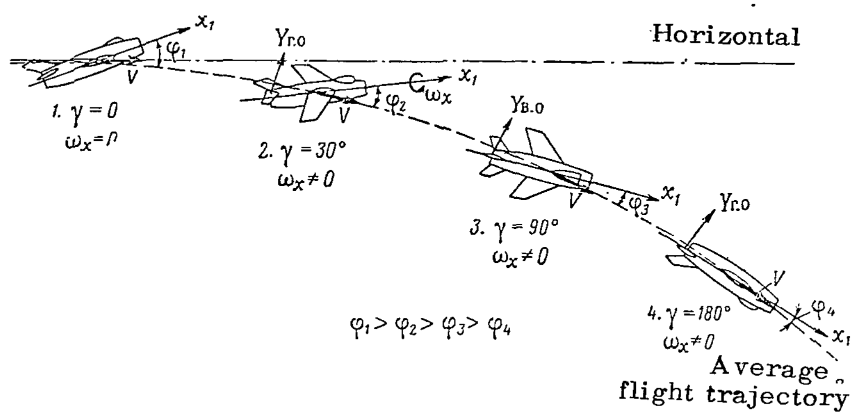

Fig. 4.43. Explanation of the Reasons for the Reduction of the Angle ϕ Between the Average Direction of the Flight Speed Vector and the Average Direction of the Longitudinal Axis of the Aircraft During the Transitional Section of Spin.

rotational moment is proportional to the speed in the first stage 157
($M_{auto} = M_x^{\omega_x}\omega_x = m_x^{\overline{\omega}_x} S l^2 \frac{\rho_H V}{4}\omega_x$), the latter will decrease more slowly with a drop in speed. This reduces the speed at which the axis of rotation approaches the direction of the flight speed vector, so that the aircraft acts more like a gyroscope, tending to retain its rotational axis in the original position in space.

CHANGES IN THE ANGLES OF ATTACK AND SIDESLIP

In view of the fact that the critical angles of attack for supersonic aircraft are relatively large (on the order of 30° and more, see for example Fig. 4.35), these aircraft begin rotating (after stalling) relative to an axis located at a relatively steep angle to the speed vector. Consequently, changes in both the angle of attack and the sideslip angle in supersonic aircraft in these regimes are very large (particularly when we take into account the change in the direction of the flight vector under the influence of gravity).

For a clear qualitative estimate of the influence of changes in the angles of attack and sideslip on the transitional section of spin, let us examine the approximate nature and reasons for the changes in these angles during a single revolution of the aircraft under the influence of the basic factors. The motion of the aircraft will be considered as taking place with the control surfaces deflected during spin (see Chapter V), and with neutral ailerons.

For the sake of simplicity (the qualitative aspect of the phenomenon will not be affected), we will not mention the damping and spiral aerodynamic moments. While the aerodynamic moments $M_y\beta = M_y^\beta\ \beta$ and $M_z\alpha = M_z^\alpha\ \alpha$ will be merely considered as having fixed sign (restoring) over the entire range of changes in α and β. We shall begin by studying yaw. In view of the above assumptions, we will consider it to be dependent upon the following five factors:

(a) The aerodynamic moment produced by tilting the rudder during spin;

(b) The gyroscopic moment of the engine rotor;

(c) Failure of the rotational axis of the aircraft to coincide with the flight speed vector, causing changes in β of the type shown in Fig. 4.42;

(d) The destabilizing inertial moment of yaw;

(e) The restoring aerodynamic moment of yaw, tending to preserve the initial sideslip.

The pitching moment can be viewed similarly. /158

The effect of these factors during one revolution of the aircraft (i.e., its revolution through 360° relative to the rotational axis, which does not coincide in the general case with the direction of the flight speed vector) in the transitional stage of spin is shown in Tables 4.1, 4.2, 4.3, and 4.4.

Even such a highly simplified analysis shows that the motion of the aircraft along the transitional section of a spin can be highly nonuniform and be accompanied by significant oscillations of the aircraft. In reality, the motion of the aircraft in this section is subject to the influence of a number of additional contradictory factors. In particular, one of the most important (as a rule) is the existence in all aircraft of a nonlinear curve of aerodynamic coefficients in the supercritical region, which considerably aggravates the irregularity of the motion and oscillation of the aircraft.

It was mentioned earlier that the greater the ratio $|\omega_x|:|\omega_y|$, the closer the rotational axis of the aircraft is located to the longitudinal axis (all other factors being equal), and therefore the more intense the oscillations and irregularity of the aircraft's motion. In other words, under comparable conditions the oscillations and irregularity of motion of an aircraft will increase as the relative value of the angular velocity of yaw decreases, as is the case (as a rule) during the transitional section of spin. It should also be pointed out that in supersonic aircraft the changes in the angular velocity of yaw during spin are much smaller than the changes in the angular velocity of roll. The latter can change very abruptly and over a relatively wide range. This is also explained by the higher mass differential in the direction of the longitudinal axis, as well as by the higher drag of the fuselage (which has greater length) and the empennage (which has a relatively greater area) in supersonic aircraft.

(b) VERTICAL SPIN

If at the beginning of the transitional section of a spin we can consider that the rotation of the aircraft takes place relative to its longitudinal axis (or nearly so), then at the end of this transitional section this assumption becomes untenable. During the process of the development of the transitional section of the spin, the rotational axis of the aircraft comes closer and closer to a direction which is tangent to the flight trajectory. In a vertical spin, the aircraft still revolves around an axis which is close to (and in some cases actually coincides with) the direction of the speed vector $\vec{V}$. In vertical spin, then, the average position of the spin axis, i.e., the vector of the angular velocity of rotation of the aircraft $\vec{\Omega}$, usually coincides with the vertical (or nearly so). The direction of the flight speed vector of an aircraft in a vertical spin usually forms a relatively small angle with the vertical. The approach of the axis of rotation to the direction of the flight speed vector produces a decrease of the oscillations and irregularity of the aircraft's motion in the regime.

The differences in the aerodynamic, design, and weight components of supersonic aircraft also cause them to exhibit significant differences in the characteristics of vertical spin, in comparison with subsonic aircraft. Comparative characteristics of vertical spin for modern and old, subsonic aircraft are shown in Fig. 4.44 (average values are shown for the angle of attack, angular velocity, and rate of descent for these aircraft in a spin). These data were obtained by analysis of a number of spins, performed beginning at initial sltitudes on the order of 5 to 8 km, with stalling at minimum speeds. The measurements were made (for example) about 20 to 25 sec after the regime began, during which time the control surfaces were completely deflected for stall and the ailerons were in a neutral position.

An analysis of the characteristic curves for vertical spin of supersonic aircraft and subsonic aircraft can begin with one of the most important parameters, the angular velocity of roll. The mean angular velocity of roll in a supersonic aircraft in a spin, as a rule, is smaller than in subsonic aircraft (see, for example, Figs. 4.21, 4.22, 4.44, and 4.45). A decrease in the angular velocity of roll in a spin makes it easier for the pilot to orient himself and reduces the stresses acting on him in this regime. For this reason, pilots sometimes say that when the initial and other conditions of a supersonic aircraft are the same, the aircraft spins "more quietly" than does a subsonic one (even though the presence of a considerable degree of nonuniformity of rotation and frequently large oscillations affect supersonic aircraft in this regime.

TABLE 4.1

INFLUENCE (DURING ONE REVOLUTION) OF VARIOUS FACTORS ON THE SIDESLIP OF AN AIRCRAFT AFTER SPIN, FOLLOWED BY ENTRANCE INTO RIGHT-HAND NORMAL SPIN

Quarters of a Revolution	Effect of Tilting the Rudder to the right	Effect of the Gyroscopic Moment	Effect of Failure of the Rotational axis to coincide with the flight speed vector	Effect of Inertial Destabilizing moment of yaw	Effect of Aerodynamic Restoring Moment of Yaw
I	Left-hand sideslip	Right-hand sideslip	Right-hand sideslip	Increase of angle β in absolute value	Impedes change in angle β.
II	" "	" "	" "	"	" "
III	" "	Left-hand sideslip	Left-hand sideslip	"	" "
IV	" "	" "	" "	"	" "

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TABLE 4.2

EFFECT DURING ONE REVOLUTION ON VARIOUS FACTORS ON THE ANGLE OF ATTACK AND THE INCLINATION OF THE LONGITUDINAL AXIS OF AN AIRCRAFT FOLLOWING A STALL WITH SUBSEQUENT ENTRANCE INTO RIGHT-HAND NORMAL SPIN

Quarters of Revolution	Effect of elevator, produced by pulling control stick backward	Effect of Gyroscopic Moment	Effect of failure of axis of rotation to coincide with flight speed vector	Effect of Inertial Destabilizing Moment of Pitch	Effect of Aerodynamic Restoring Moment of Pitch
I	Increase in angle α, nose of aircraft goes up	Increase in angle α, nose of aircraft goes up	Increase in angle α, nose of aircraft goes down	Increase of angle α in absolute value, nose of aircraft goes up	Impedes change in angle α (with an increase in α the nose rises, with a decrease in α the nose falls).
II	Increase in angle α, nose of aircraft goes down	Increase in angle α, nose of aircraft goes down	"	Increase of angle α in absolute value, nose of aircraft goes down.	Impedes change in angle α (with an increase in α the nose falls, with a decrease in α the nose rises).
III	"	"	Increase in angle α, nose of aircraft goes up	"	"
IV	Increase in angle α, nose of aircraft goes up	Increase in angle α, nose of aircraft goes up	"	Increase of angle α in absolute value, nose of aircraft goes up.	Impedes change in angle α (with an increase in α the nose rises, with a decrease in α the nose falls).

TABLE 4.3.

EFFECT OF VARIOUS FACTORS ON SIDESLIP DURING ONE REVOLUTION OF AN AIRCRAFT FOLLOWING A STALL WITH SUBSEQUENT ENTRANCE INTO LEFT-HAND NORMAL SPIN

Quarters of Revolution	Effect of Tilting the Rudder to the left	Effect of Gyroscopic Moment	Effect of Failure of Rotational Axis to coincide with flight speed vector	Effect of Inertial Destabilizing Moment of yaw	Effect of Aerodynamic Restoring Moment of Yaw
I	Right-hand sideslip	Right-hand sideslip	Left-hand sideslip	Increase in angle β in absolute value	Impedes change in angle
II	"	"	"	"	"
III	"	Left-hand sideslip	Right-hand sideslip	"	"
IV	"	"	"	"	"

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TABLE 4.4

CHANGE (DURING ONE REVOLUTION) OF THE INFLUENCE OF VARIOUS FACTORS ON THE ANGLE OF ATTACK AND THE SLOPE OF THE LONGITUDINAL AXIS OF THE AIRCRAFT FOLLOWING STALL WITH SUBSEQUENT ENTRY INTO LEFT-HAND NORMAL SPIN

Quarters of Revolution	Effect of elevator, produced by pulling control stick backward	Effect of Gyroscopic Moment	Effect of failure of axis of rotation to coincide with flight speed vector	Effect of Inertial Destabilizing Moment of Pitch	Effect of Aerodynamic Restoring Moment of Pitch
I	Increase in angle α; nose of aircraft rises.	Decrease in angle α; nose of aircraft falls.	Decrease in angle α; nose of aircraft falls	Increase of angle α in absolute value; nose of aircraft falls.	Impedes change in angle α (when α increases, the nose rises; when angle α decreases, the nose falls).
II	Increase in angle α; nose of aircraft falls.	Decrease in angle α; nose of aircraft rises.	"	Increase of angle α in absolute value; nose of aircraft falls.	Impedes change in angle α (when α increases, the nose falls; when α decreases, the nose rises).
III	"	"	Increase in angle α; nose of aircraft rises.	"	"
IV	Increase in angle α; nose of aircraft rises.	Decrease in angle α; nose of aircraft falls.	"	Increase of angle α in absolute value; nose of aircraft rises.	Impedes change in angle α (when α increases, the nose rises; when α decreases, the nose falls.

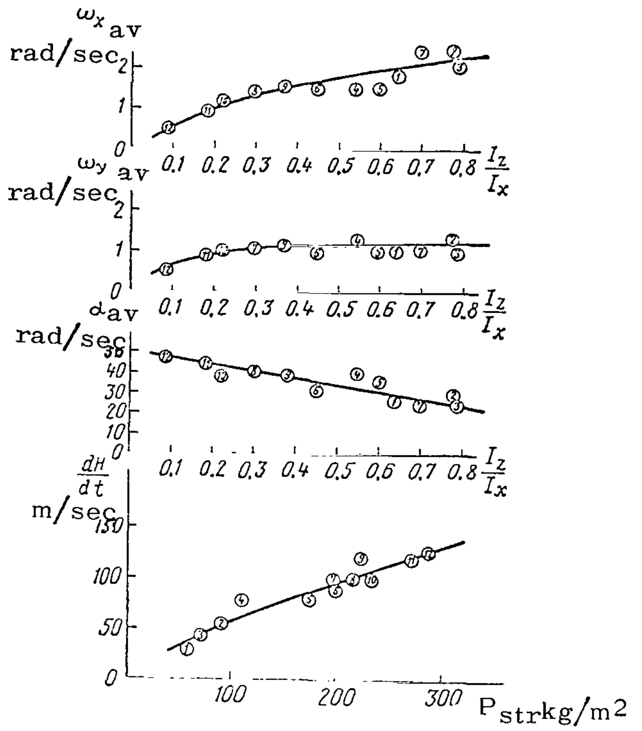

Fig. 4.44. Changes in Spin Characteristics as a Function of the Mass Distribution over the Aircraft and the Wing Loading. /161

1 = R-5	7 = La - 11
2 = I-14	8 = MIG-15
3 = Ut-2	9 = MIG-15 UTI
4 = LaGG-3	10 = MIG-17
5 = R-39	11 = MIG-19
6 = MIG-9	12 = MIG-21

The decrease in the mean angular velocity of rotation of the aircraft in a spin is explained mainly by the increase in the mass distribution along its longitudinal axis, as well as by an increase in the length of the fuselage, which increases the drag produced by the fuselage itself and the empennage.

The average angle of attack of supersonic aircraft in a spin α_{av} is usually much greater than in subsonic aircraft (see Fig. 4.44). This is due mainly to the increase in the inertial moment of pitch, produced by centrifugal forces acting on the masses distributed along the length of the fuselage (see Fig. 4.17), as well as the increase in the critical angles of attack in supersonic aircraft. It is clear from Fig. 4.44 that the average angles of attack in normal spin for modern supersonic aircraft under the conditions with which we are dealing, fall within the limits of 50° $\pm$ (5 to 10°), varying mainly between 45° and 50°. In the case of subsonic aircraft used in World War II, the average values for the angle of attack in normal spin were usually on the order of 28 to 35°, i.e., much smaller.

The rate of descent for supersonic aircraft in a spin is much greater than for subsonic aircraft (see Fig. 4.44). This is due mainly to the increased wing loading and the deterioration of the left of the latter. The drop in the mean angular velocity of rotation ω_{av} means that the average radius of spin r_{av} in supersonic

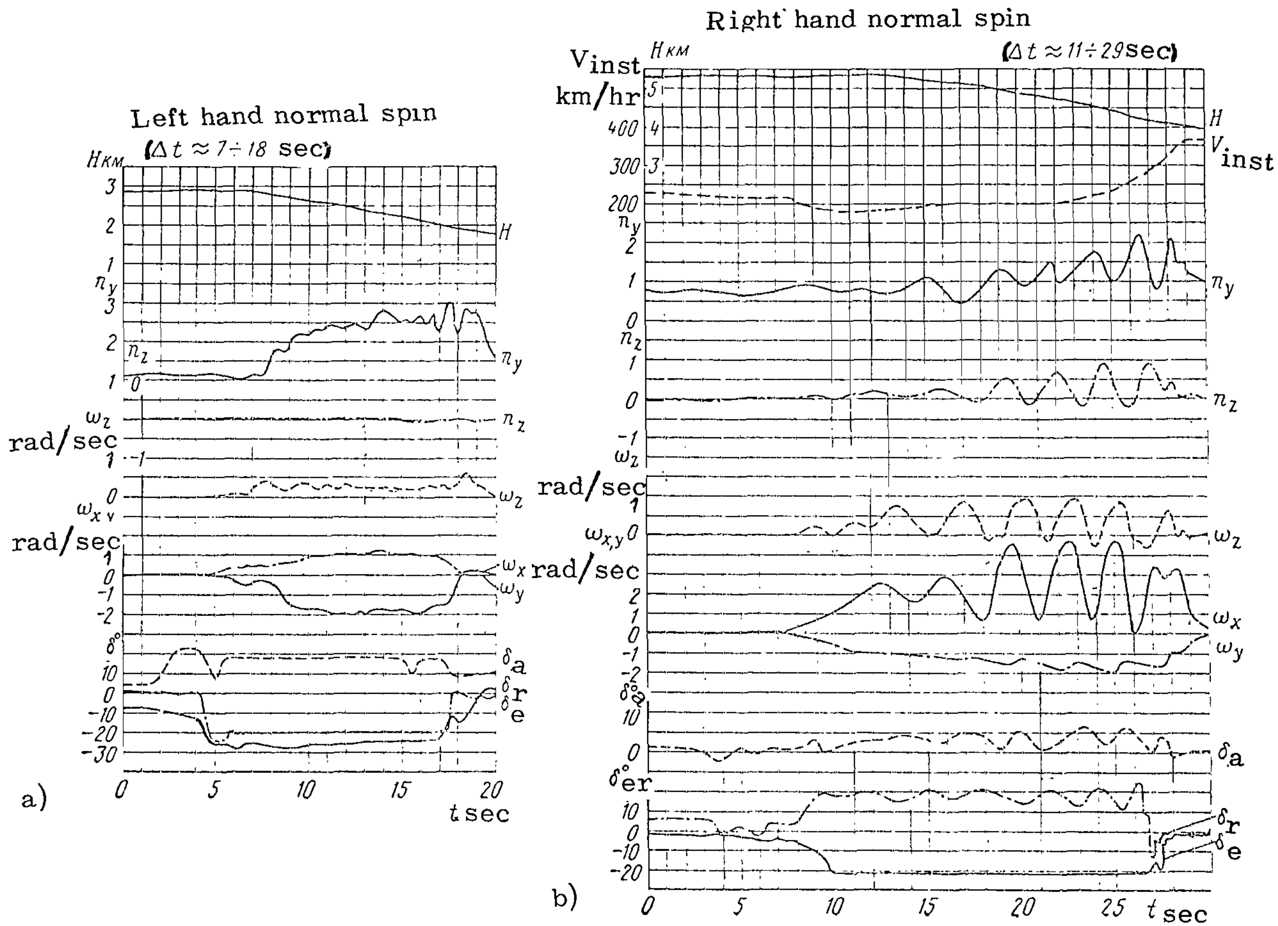

Fig. 4.45. Examples of Spin in Old Subsonic Aircraft: (a) R-5, (b) La-11.

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aircraft is greater. This is because the value of the average spin radius (radius of the spiral spin) is given by the following approximate relationship: /163

$$r_{av} = \frac{g}{\omega_{av}^2 \tan\alpha_{av}}$$

Despite the increase in tan α_{av}, the value r_{av} also increases, since a drop in the mean angular rotational velocity, which appears squared in this equation, is predominant (see, for example, Fig. 4.44).

(c) DIVERSITY AND INSTABILITY OF SPIN CHARACTERISTICS

The expansion of operating flight ranges, Mach numbers and instrumental flight speeds, increase in the effectivness of control surfaces and ailerons at subsonic speeds, transition to wings and empennages with greater sweepback angles, an increase in the thrust-weight ratio (especially involving an increase in the effect of the gyroscopic moments of the engine rotors), etc., have all contributed to the fact that the spin characteristics of supersonic aircraft are much more diverse and unstable than is the case for subsonic aircraft. A given supersonic aircraft may exhibit spin characteristics which are quite different, depending on the initial condi- /164
tions for entrance into spin, the duration of the regime, the position of the control surfaces and ailerons during spins, etc. As a rule, supersonic aircraft are subject to more irregular motion and greater oscillation during spin periods.

IRREGULARITY AND INSTABILITY OF SPIN

Irregularity and instability (changes with time) in the regime, especially at high flight altitudes, are one of the most important features of spin in supersonic aircraft. The main reasons for the occurence of these phenomena were mentioned previously in the discussion of the transitional section of spin: nonlinearity of the aerodynamic characteristics, the action of the gyroscopic moment of the engine, and the effect of failure of the rotational axis of the aircraft to coincide with the direction of the flight speed vector (the latter occurs mainly during the transitional section of spin).

Modern supersonic aircraft are most often prone to regimes of unstable normal spin, which occur very irregularly and involve considerable oscillation of the aircraft, often with interruptions, and frequently changing direction of rotation. Occasionally, regimes of unstable spin are observed which take place with considerable variation of parameters and terminate with the aircraft emerging from spins by itself (with the control surfaces tilted with spin and neutral ailerons). These regimes are of two types: falling along a spiral trajectory like a leaf, and increasing in oscillation. The occurence of these regimes is caused primarily by the presence of a sharply pronounced nonlinear pattern of aerodynamic coefficients and especially by those coefficients of aerodynamic moments of yaw and pitch which depend on the angles of attack and sideslip.

REASONS FOR WOBBLE AND FLUTTER

Spin which involves wobbling of the aircraft (see Fig. 4.21) is the result of superposition of two forms of oscillation, occurring at very similar frequencies. Studies have shown that the occurrence of such oscillations in an aircraft is possible under the influence of a nonlinear pattern of coefficients of aerodynamic moments of yaw and pitch (of the type shown in Fig. 4.46).

The interaction of the aerodynamic and inertial moments (in particular) and forces with essentially nonlinear relationships between the changes in the aerodynamic coefficients with angle of attack and sideslip, can lead to very pronounced changes in the motion parameters of an aircraft in this regime. This interaction can produce periodic movements of the aircraft at large supercritical and relatively low subcritical angles of attack, with /16
periodic spontaneous interruptions in this rotation, sometimes even involving transition of the aircraft from spin in one direction to spin in the opposite direction, thus causing the aircraft to flutter downward like a leaf. Studies have shown that the spin regime which takes place under these conditions, with the aircraft falling downward like a leaf along a spiral trajectory (see Fig. 4.22), can occur particularly when nonlinear relationships $m_z = f(\alpha)$ and $m_y = F(\beta)$, of the type shown in Fig. 4.46 and 4.47 are in existence.

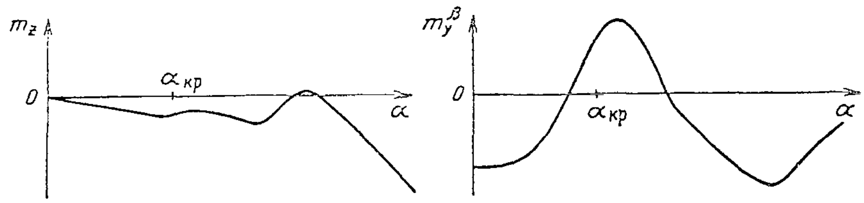

Fig. 4.46. Examples of Possible Nonlinear Curves $m_z = f(\alpha)$ and $m_y^\beta = \phi(\alpha)$.

INCREASE IN FORCE AND DIVERSITY OF REGIME

Supersonic aircraft characteristically show a considerable increase in the average values of force in relatively prolonged regimes of unstable spin. Thus, for example, in the spin shown in Fig. 4.24, the average normal force $n_{y_{av}}$ begins to increase about 16 sec after the beginning of the regime (i.e., approximately at the end of the transitional section of spin), and after 25-30 seconds following the beginning of the regime, the value $n_{y_{av}}$ exceeds $n_y = 2$. Investigations have shown that such an increase in

$n_{y_{av}}$ can occur at both fixed and decreasing average angles of attack during spin. This is due to an increase in the instrument speed (reference pressures) during the regime.

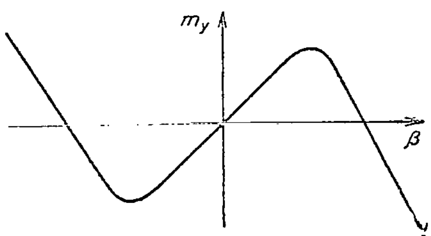

Fig. 4.47. Example of the nonlinear relationship $m_y = F(\beta)$.

The diversity of spin regime in supersonic aircraft is explained in particular by the fact that a given aircraft can show many different forms and varieties of spin, depending on a number of factors. Thus, for example, as shown in Fig. 4.22, a regime of unstable spin which occurs with the aircraft fluttering downward along a trajectory like a leaf, and the regime of very stable uniform spin, shown in Fig. 4.29, were obtained during the flight of a single aircraft. /166

Supersonic aircraft frequently have various external structures (supports for fuel tanks and jet engines, etc.). The presence of these structures further increases the diversities of regimes, and often causes an increase in the oscillations of the aircraft during spin, thus increasing the irregularity of its motion in this regime.

4.4 Engine Operation During Spin

Modern supersonic aircraft as a rule are equipped with turbojet engines (TJE). The operating regime of a TJE can have a considerable influence on the spin characteristics. In a spin which occurs with the engine in operation, the aircraft is subject to other forces and moments besides those discussed above (see section 4.1): the thrust of the engine, the moment consisting of the thrust of the engine relative to the center of gravity of the aircraft, and the gyroscopic moment. In addition, considerable influence on the characteristics of spin can be exerted by the normal and side forces which arise as a result of the rotation of the incident airflow upon entering the air intake. Significant influence can also be produced by a change in the nature of the airflow over the nose section of the fuselage (when the air intakes of the aircraft are located in the nose), depending on the operating regime of the engine and the changes in the effectiveness of the empennage under the influence of the compressing effect caused by the flow of air from the exhaust nozzle of the jet engine. Hence, changes in the operating regime of the engine can change the characteristics of the spin of the aircraft considerably.

On the other hand, the operating conditions of the engine itself, (more precisely, the entire power unit of the aircraft as a whole) differs markedly during spin from the operating conditions of the same engine under all other (operational) flight regimes. In other words, spin involves a mutual effect of the operational effect of the engine on the spin characteristics, and vice versa. Let us look at both of these problems briefly.

(a) OPERATIONAL CONDITIONS FOR AN ENGINE DURING SPIN

When a supersonic aircraft is spinning, the TJE operates in non-calculated conditions in which it is especially possible that the engine will stop as a result of pumping breakdowns, while the piston engines of subsonic aircraft generally operate without any difficulty. Sometimes, it is true, there were cases when the piston engines were choked because the propeller of the aircraft became too "heavy" for the motor, operating in a low-gas regime. However, this was not dangerous to the engine itself (no overloads or other unpleasant undesired effects were imposed on it). We can /167 therefore state that in all cases when an aircraft fell into a spin, the piston engines required practically no additional attention from the pilot.

The first jet aircraft (for example, the MIG 15, MIG 17, etc.) were equipped with engines using centrifigal compressors, which are less sensitive to conditions at the intake than are engines with axial compressors. Therefore, controlling the engine during spin did not pose any particular difficulties for the pilot in such aircraft any more than it did in aircraft with piston engines. Engine failures in aircraft fitted with centrifigal compressors were not observed in spins. These engines operated normally in stalls, spins, and when emerging from spins.

The transition to engines with axial compressors led to a deterioration of the operational stability and the appearance of engine failures during spins. This was because the engines with high-pressure axial compressors which are used in modern supersonic aircraft are very sensitive to irregularities in the pressure field at the engine intake. However, with the air intake used on modern supersonic aircraft, flight at large angles of attack and sideslip, and also at large angular rotational velocities and angular accelerations (with considerable oscillations) cause the aircraft to show considerable irregularity of flow into the intakes of the power plant. As a result, the pumping of the engine may be interrupted. In other words, the conditions for engine operation during spin, in the case of a supersonic aircraft, are less favorable (especially at high altitudes) than was the case in subsonic aircraft. Such a situation requires that the pilot give increased attention to the operation of the engine when a supersonic aircraft enters a spin.

In the case of supersonic aircraft with TJE's, which have stalled as a result of interruption of pumping without any further dangerous increase in gas temperature (which could lead to overheating of the engine), the pilot must see to it that the engine is started in time and correctly after the aircraft has emerged from the spin. This means in particular that the aircraft must first be speeded up to ensure reliable starting of the engines and that the pilot must make certain other additional moves after the aircraft has emerged from spin.

In aircraft with TJE's, in which an intense temperature increase of the gases can occur after the engine has cut out, the pilot must devote additional time to dealing with the problem. When the spin begins, the pilot must immediately shut off the engine (switch it to a regime of autorotation). If he does not do so, the pilot may be unable to follow the changes in the gas temperature in the engine during pumping (i.e., he may not notice when the moment of pumping occurs, having unexpectedly fallen into such a complex nonoperational flight regime as spin) so that the TJE may overheat.

(b) EFFECT OF OPERATIONAL REGIME OF THE ENGINE ON SPIN CHARACTERISTICS /168

On the whole, the effect of the operational regime of the motor on the spin characteristics in jet aircraft is much smaller than in aircraft with piston engines. This is explained by the absence of reactive moments and a smaller effect of the gyroscopic moment in aircraft with TJE's as well as by the absence of any blowing of currents from the propeller over the surface of the aircraft, which in some cases has played a crucial role in the spin of aircraft with piston engines. The influence of the operational regime of the engine on the characteristics of spin in supersonic aircraft is explained primarily by the gyroscopic moment which is formed by the rotating TJE rotor.

In modern supersonic aircraft the influence of the gyroscopic moment of the engine is greater than the older jet aircraft. The reason for this is to be found in the increased absolute values of $J_p\omega_p$ in the TJE's of supersonic aircraft, related to an increase in their thrust-weight ratio. The nature of the changes in the absolute value of $J_p\omega_p$ in Soviet-built aircraft of the period following World War II can be determined by examining the aircraft designed by A. I. Mikoyan and M. I. Gurevich. Table 4.5 shows the parameters which characterize the values of the gyroscopic moment of the TJE rotor in these aircraft, with values of ω_p which correspond to the revolutions in autorotation of their engines during spin at altitudes on the order of 8-10 km.

Chapter I contained a formula for determining the magnitude and direction of the action of the gyroscopic moment. This relationship can be expressed more conveniently by means of the following simple rule.

To determine the direction of the additional motion of the aircraft nose, which is produced by the action of the gyroscopic moment, we must turn the arrow showing the direction of the forced motion of the nose of the aircraft (as seen from the cockpit)

TABLE 4.5.
CHARACTERISTICS OF GYROSCOPIC MOMENTS OF ENGINES IN SOVIET-BUILT AIRCRAFT DESIGNED BY A.I. MIKOYAN AND M.I. GUREVICH.

Parameter of gyroscopic moment \ Aircraft	MIG-9	MIG-15	MIG-17	MIG-19	MIG-21
$\lvert J_p\omega_p\rvert$ kg•m/sec	44.4	59.6	105	192	255
$\frac{\lvert J_p\omega_p\rvert}{J_y}$ rad/sec	0.0280	0.0298	0.0429	0.0433	0.0481
$\frac{\lvert J_p\omega_p\rvert}{J_z}$ rad/sec	0.0326	0.0331	0.0488	0.0512	0.0517

through 90° from the direction of rotation of the engine rotor. Turning the arrow in this way shows the direction of motion of the aircraft nose under the influence of the gyroscopic moment. Fig. 4.48 shows a diagram which clearly illustrates this rule. /169

It is particularly clear from this drawing that with left-hand[1] rotation of the engine rotor ($\omega p < 0$), during left-hand spin the nose of the aircraft will turn to the left ($\omega_y > 0$), thus causing an additional motion of the aircraft so that it enters a dive ($\omega_z < 0$) under the influence of the gyroscopic moment; see Fig. 4.48a. This produces a drop in the average angle of attack. In this case, dropping of the aircraft nose (see Fig. 4.48,c) causes a drop in the angular velocity of yaw, which in turn reduces the action of the gyroscopic moment directed toward lowering the nose of the aircraft, etc. Hence, the action of the gyroscopic moment in this spin regime reduces the average angle of attack and the angular velocity of roll: their variations, on the other hand, increase. An increase in the oscillations of the angular velocity of yaw cause an increase in the oscillation of the angle of sideslip.

The nature of the influence of the gyroscopic moment on the characteristics of spin are uniform and exist to various degrees in all aircraft. The greater the absolute value of oscillations in the angular velocity of rotation of an aircraft during spin (with constant $J p \omega p$), the more strong and abrupt the influence of the gyroscopic moment will be. These results are supported by data from flight tests of modern aircraft. It is clear from the example shown on Fig. 4.49 that in the supersonic aircraft in question, which was fitted with an engine with left-hand rotation, the oscillations are much greater in left-hand spin (Δ_{nz} max ≈ 1), than in right-hand spin (Δ_{nz} max ≈ 0.1).

[1] Left-hand rotation takes place counter-clockwise as viewed by an observer seated in the cockpit and looking forward (GOFT 1630-46).

The nature of the change in the average values α_{av} and ω_{yav} in right-hand and left-hand spin also confirm the statements made above. An analysis of the results of flight tests for different aircraft during spin shows that in supersonic aircraft, the difference between right-hand and left-hand spins (the influence of the gyroscopic moment of the TJE rotor) are much greater than in subsonic jet aircraft. This conclusion is also supported by a comparison of the values of maximum oscillations in lateral force and angular velocity of roll in right-hand and left-hand spins for the following aircraft: MIG-9, MIG-15, MIG-17, MIG-19, and MIG-21, as shown in Table 4.6.

TABLE 4.6.

Spin parameter \ Aircraft	MIG-9	MIG-15	MIG-17	MIG-19	MIG-21
$\frac{(\Delta\omega_y)\ \text{left}}{(\Delta\omega_y)\ \text{right}}$	1.09	1.11	1.25	1.35	1.5
$\frac{(\Delta n_z)\ \text{left}}{(\Delta n_z)\ \text{right}}$	1.1	1.26	1.4	1.65	1.77

4.5 Influence of Initial Altitudes and Flight Speeds

(a) INFLUENCE OF ALTITUDE

The increase in the flight altitude of supersonic aircraft, whose dynamic ceilings at the present time approach 50 km, means that there is a possibility of stalling and spin at very high altitudes.

In particular, according to reports in the foreign press during December, 1964, the famous test pilot and director of the test pilot school at Edwards Air Force Base in the United States, Charles Yeager, fell into a spin at an altitude at 30.8 km. This took place in a Lockheed NF-104A, while he was making a preparatory flight before attempting to set a new world record for altitude (flight at the dynamic ceiling). After repeated unsuccessful attempts to pull the aircraft out of the spin, Yeager was forced to bail out.

The initial altitude can have considerable influence on the spin characteristics (see Fig. 4.50). The tendancy of supersonic aircraft to go into a spin considerably increases the irregularity and instability of this regime at high altitudes. In the event of stalling at high altitudes, the oscillations increase considerably and the rate at which the aircraft falls into a spin also is

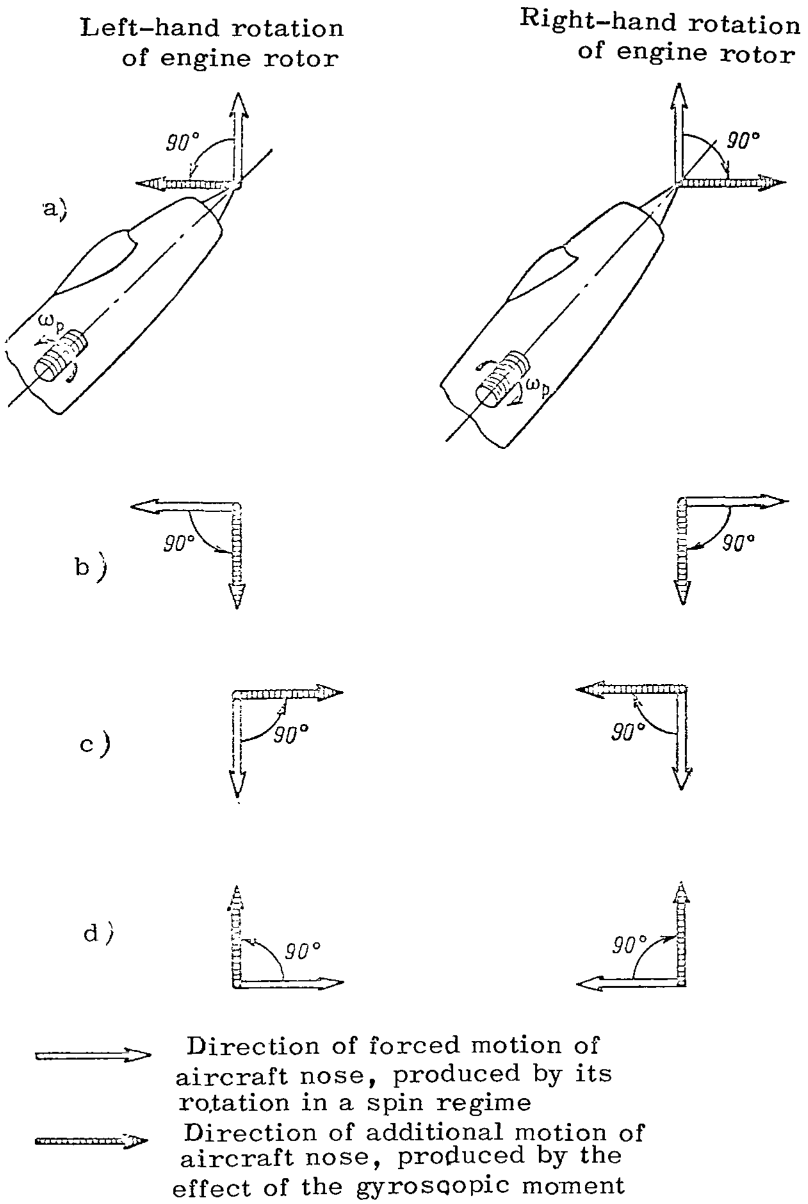

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Fig. 4.48. Diagram Illustrating the Direction of the Action of the Gyroscopic Moment of the Engine Rotor.

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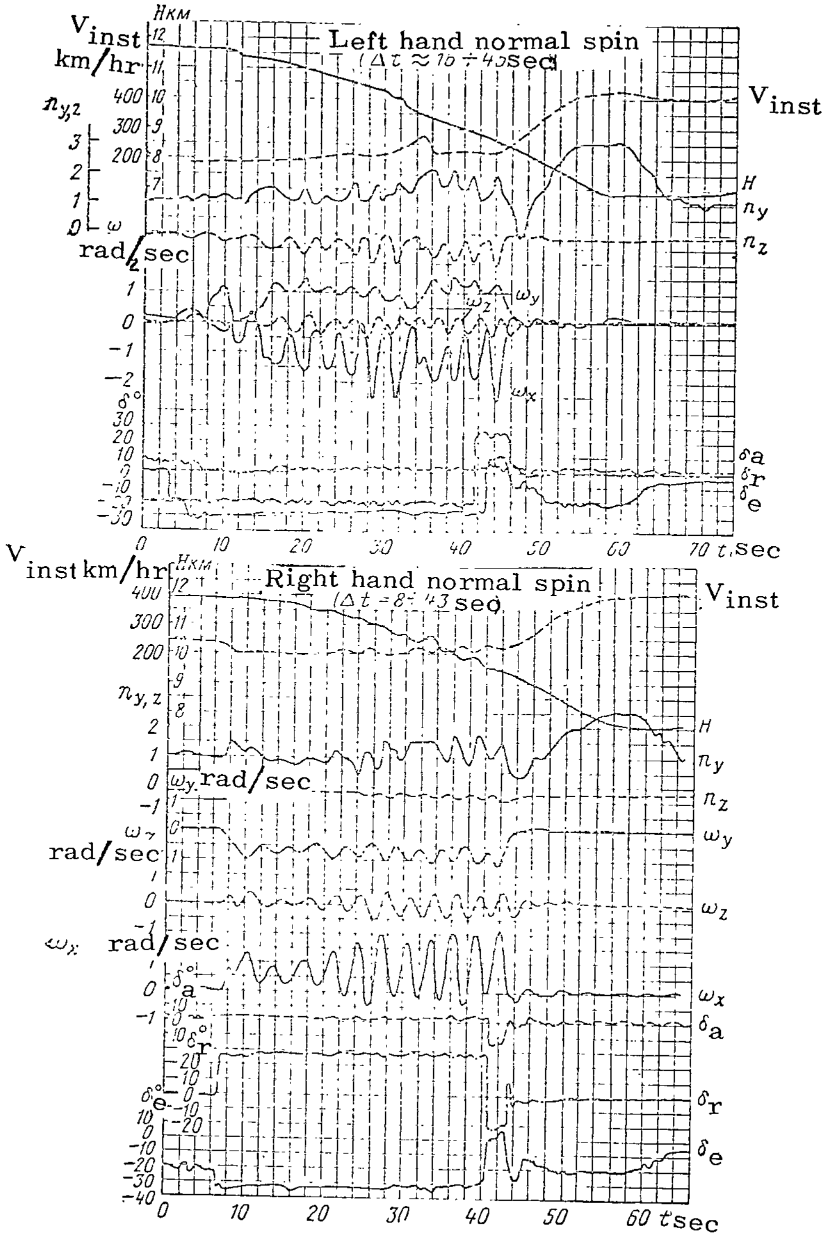

Fig. 4.49. Influence of Gyroscopic Moment on the Characteristics of Spin in a Supersonic Aircraft.

greater; the difference between the regimes of right-hand and left-hand spin increases as well (see Fig. 4.51). The aircraft spins with very little stability or even with no stability at all. Frequently, the motion of the aircraft in such a spin is very irregular. The periodic rotation of the aircraft speeds up and slows down and sometimes stops briefly. /172

The increase in the rate of descent of the aircraft during the spin, which accompanies an increase in the flight altitudes, is caused mainly by the fact that with a constant value of instrument speed, (constant velocity head), the actual flight speed increases with increasing altitude. An increase in the difference between the regimes of right-hand and left-hand spin is explained by the increase in the influence of the gyroscopic moment of the TJE. The latter is caused in turn by an increase in the number of revolutions during autorotation at high altitudes (if spin occurs with the engine not operating). /174

In the case of spin which takes place after stalling at high altitudes, there are (as a rule) very considerable oscillations in the angular velocities, the angles of slope of the aircraft, the angles of attack and sideslip, as well as the corresponding load factors. In the case of spin at high altitudes, the relationship of the average absolute values of the angular velocities of yaw and roll are usually much smaller than is the case at lower altitudes. Longitudinal oscillation of the aircraft may be accompanied by changes in the pitch angle which are so great that the nose of the aircraft (in a regime of normal spin) periodically rises above the horizon and then dips down until it is in a vertical position (or nearly so). The roll angle can then change within limits of $\pm$ 180° and more; the aircraft may periodically go on its back.

All of these factors considerably hamper visability of the horizon (even under excellent meteorlogical conditions, the pilot usually sees only a part of the line of the horizon), and considerably complicates the orientation of the pilot, i.e., it is much more difficult for him to have a correct idea of the nature of the motion of the aircraft and its position in space. The reduction in the stability of the aircraft motion in this regime and the increase in the irregularity of the rotation increases (or creates, if it is absent in spin at relatively low altitudes) a tendency towards a spontaneous periodic pause in the aircraft rotation (ω_y = 0) and even causes it to make a transition from normal spin in one direction to normal spin in the other direction, from right-hand to left-hand, and vice versa (with fixed position of the controls).

A reduction in the stability of the aircraft motion and an increase in the longitudinal oscillation (changes in the amplitudes of the angle of attack) increases the tendency of the aircraft to make a spontaneous transition from normal to inverted spin and vice versa with the control surfaces in the same position as they were when the spin began.

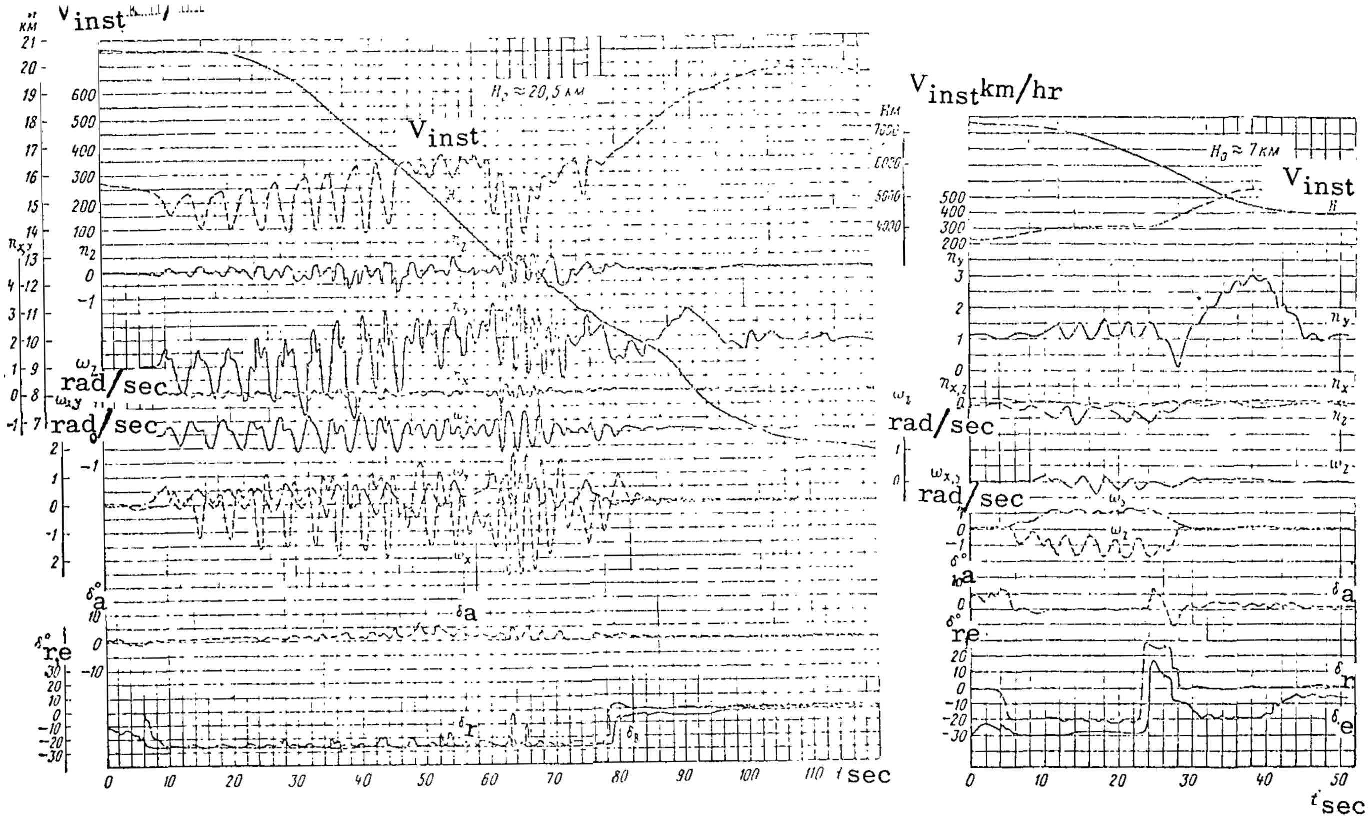

Fig. 4.50. Examples of Spin in Supersonic Aircraft Following Stalling at Initial Altitude $H_0 \approx 20.5$ km and $H_0 \approx 7$ km.

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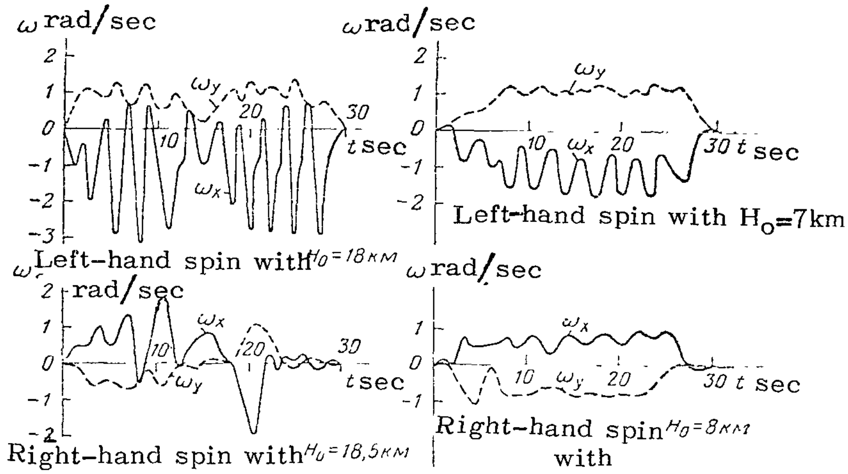

Fig. 4.51. Changes in the Angular Velocities of Rotation for Supersonic Aircraft in Right-Hand and Left-Hand Normal Spin, Following Stalling at High and Relatively Low Altitudes.

As a rule, the motion of supersonic aircraft in normal spin at high altitudes involves flutter, with the aircraft falling downward along a spiral trajectory like a leaf. This may involve wobble, or more rarely, increasing oscillation of the aircraft in the regime. All of this means that even a very experienced pilot will find it difficult under such conditions to differentiate a right-hand spin from a left-hand spin and normal flight from inverted flight. /17

Figure 4.52 shows an example of the change of characteristics of spin for a supersonic aircraft as a function of altitude. The data shown in Fig. 4.52 were recorded 20-25 seconds after the aircraft stalled at a minimum velocity ($n_{y0} \approx 1$) with the control surfaces tilted with stall and the ailerons in a neutral position.

Ground and flight tests show that such a considerable influence of the initial altitude on the spin characteristics is explained not only by the actual reduction in the air density at high altitudes, but also (and primarily) by the influence of changes in parameters such as the Mach and Reynolds numbers. Changes in the latter affect the aerodynamic coefficients and their derivatives according to the angles of attack and sideslip, thus producing a considerable intensification of the nonlinear pattern for these relationships. A drop in the aerodynamic damping and an increase in the relative value of the inertial moments (especially yaw and pitch) at high altitudes also produces an increase in the oscillation and irregularity of the aircraft's motion during the spin. With a fixed initial velocity head (stall at $V_{inst} \approx V_{min}$), the duration and length of the transitional section of spin increased considerably with an increase in the initial flight altitude, which can be explained primarily by an increase in the initial actual flight speed (an increase in the kinetic energy of the aircraft). However, even with a fixed initial actual stall speed and an increase in the stalling altitude, the duration and extent of the transitional section of spin increased. This is produced by a reduction in the aerodynamic forces (primarily drag) and moments (especially the moment acting on the empennage and causing the nose of the aircraft to drop) with a decrease in the air density.

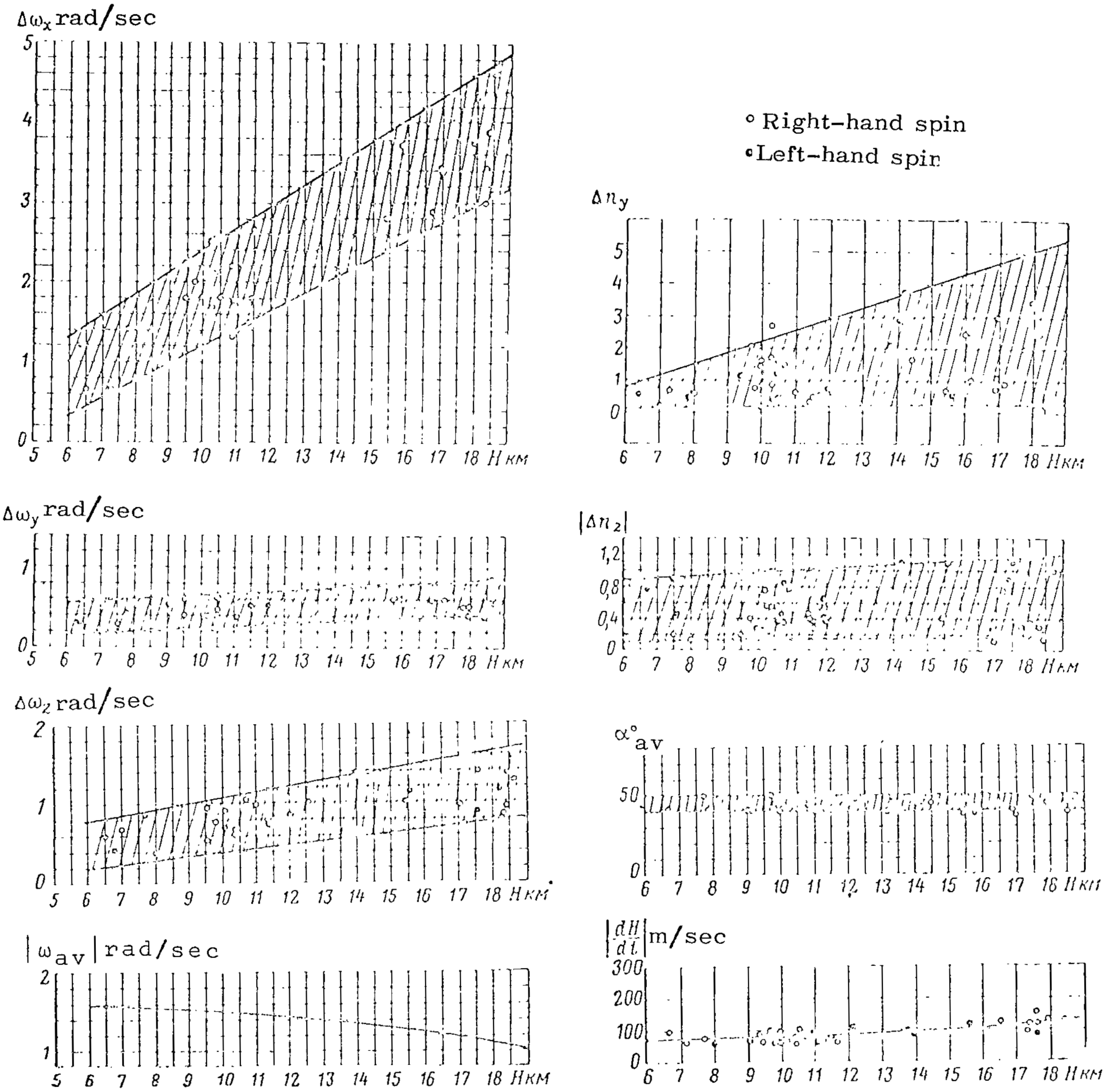

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Fig. 4.52. Change in Spin Characteristics for a Supersonic Aircraft with Altitude

(b) CHANGES IN VELOCITY DURING THE REGIME /17

Due to the considerable extent of the transitional section of spin in the case of supersonic aircraft, particularly at high initial flight altitudes, the following changes in the flight speed during spin generally occur after stalling at an initial regime of straight-line flight at $V_{inst} \approx V_{min}$. At the beginning of the transitional section, when the drag on the aircraft increases considerably with transition to supercritical angles of attack, and the axis of spin is still tilted only slightly with respect to the horizon, the instrument and actual flight speed will decrease. However, when the spin axis turns relative to the horizon so that the projection of the aircraft weight along a tangent to the trajectory of the aircraft's center of gravity becomes greater than the drag, the instrument flight speed will begin to increase. This usually results in an increase in the actual flight speed, but to a lesser extent since the air density increases as the aircraft falls (usually the actual speed increases to a value which is smaller than its initial value). The occurrence of a stable (strictly speaking, quasi-stable) regime of vertical spin means that the instrument flight speed becomes practically constant. With a constant instrument speed, the actual flight speed decreases constantly as the aircraft falls.

In the case of stalling at the dynamic ceiling, when the instrument flight speed $V_{inst} \ll V_{min}$, the aircraft speeds up considerably during the transitional section of spin. This is caused by the fact that the average value of the flight speed of the aircraft in a regime of vertical spin, in which a balance of forces is achieved (even approximately), acting on the aircraft in the vertical direction, can be considered close to the value V_{min} in a first approximation for the sake of simplicity. The latter condition, strictly speaking, would be permissible only when the corresponding components of the resultant aerodynamic forces, gravity and weight, acting on the aircraft during stall and in the regime of vertical spin, were equal, i.e., if in both cases the values of the projections of these forces on the vertical were the same. This feature causes a considerable increase of the unstable transitional section of the spin in the case of stalling at the dynamic ceiling with a low initial flight speed.

(c) EFFECTIVE INITIAL VELOCITY AND MACH NUMBER

The effect of the magnitude of the initial instrument flight speed on the spin regime has been mentioned before (see Fig. 3.16). As we have already pointed out, even the increase in the initial instrument speed (without taking into account the changes in aerodynamic coefficients which are involved due to the influence of the compressibility and viscosity of the air), this leads to a considerable increase in the abruptness of stall. However, this increases the irregularity of motion and oscillation of the aircraft during spin, owing to an increase in the aerodynamic forces

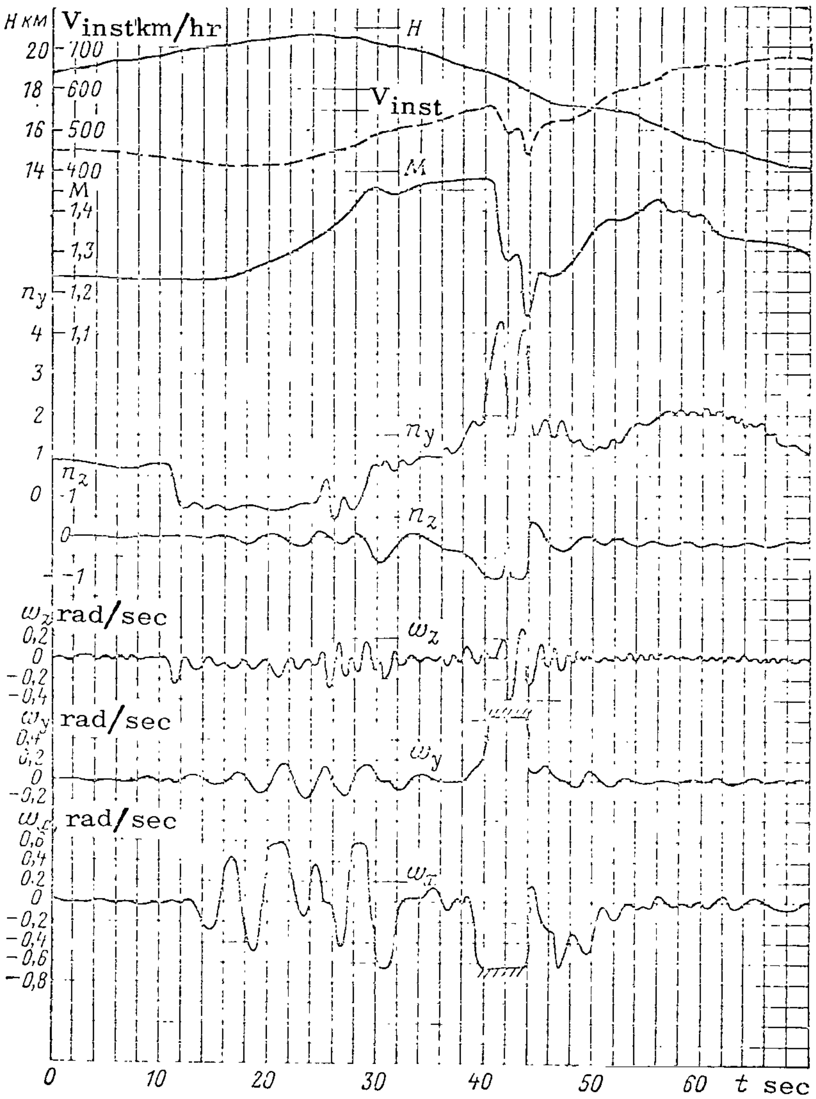

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Fig. 4.53. Spin of an Aircraft; Entrance ($t \approx 39$ sec) and Emergence ($t \approx 44$ sec) Occurred at Supersonic Speeds.

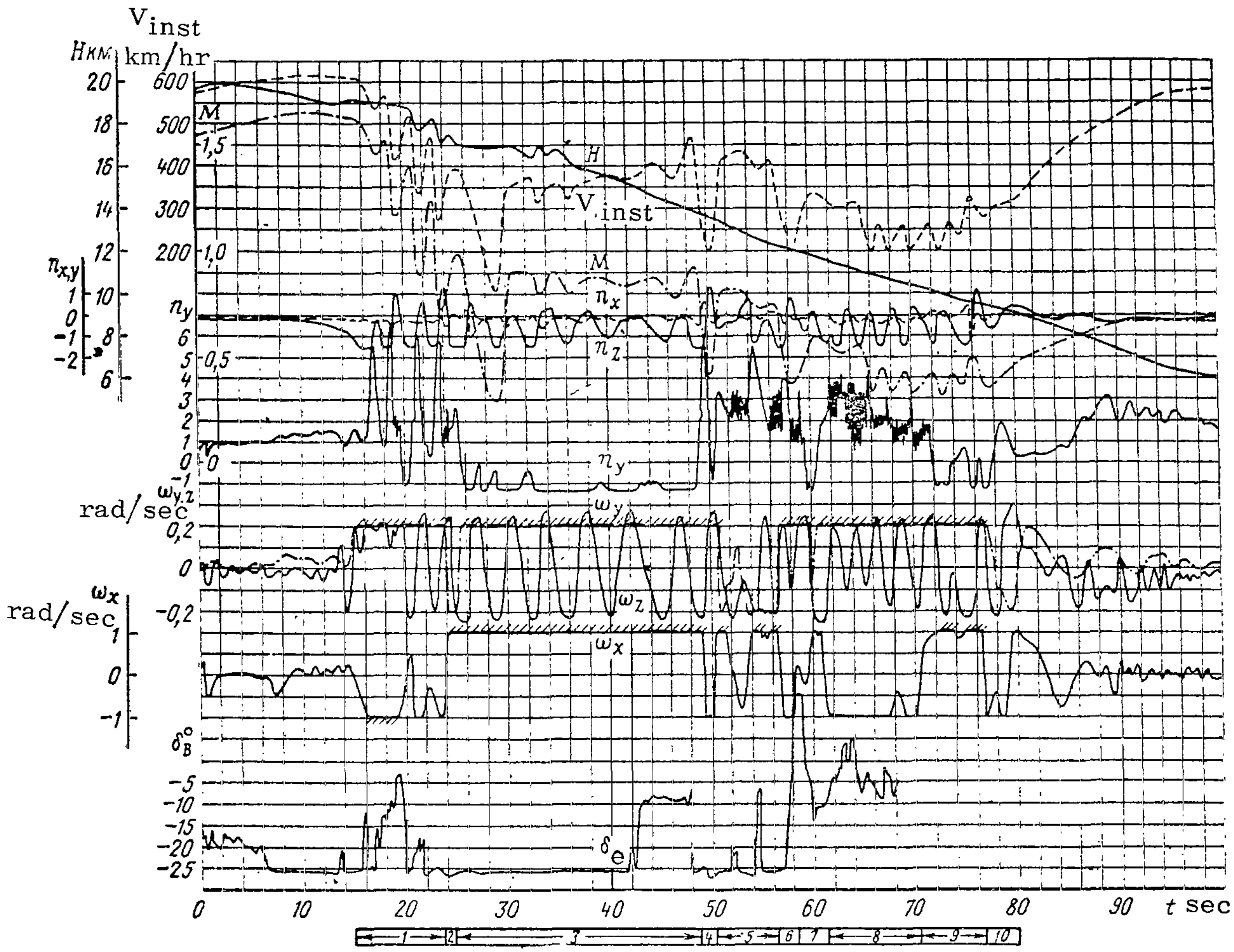

Fig. 4.54. Spin of an Aircraft Following Stalling at an Initial Speed of Mach ≈ 1.65 During Which Repeated Changes in the Nature of the Regime Occurred: (1) Left-Hand Normal Spin; (2) Right-Hand Normal Spin; (3) Left-Hand Inverted Spin; (4) Left-Hand Normal Spin; (5) Right-Hand Normal Spin; (6) Left-Hand Normal Spin; (7) Left-Hand Inverted Spin; (8) Left-Hand Normal Spin; (9) Left-Hand Inverted Spin; (10) Right-Hand Inverted Spin.

and moments, and an increase in the velocity head as well. With a transition to spin following stalling at a high instrument speed, there are considerable longitudinal and lateral oscillations of the aircraft, which rapidly die out as the flight speed drops. Since an increase in the initial instrument flight speed causes an increase in the Mach and Reynolds numbers, the effect mentioned is intensified still further. /181

Particularly large oscillations and very abrupt and nonuniform motion of the aircraft in spin are encountered in the case of stalling at high supersonic speed (Mach numbers), as we can see, e.g., from Figures 4.53 and 4.54. This causes a considerable increase in the transitional section of the spin, and a large moment of autorotation (high instrument flight speed) means that the rotation of the aircraft immediately after stalling may be very intense. Following stalling at supersonic Mach numbers in the case of relatively short spin regimes, it is possible for supersonic speeds to be retained; the aircraft may even emerge from the spin (see Fig. 4.53).

Following stalling at high supersonic speeds, there is a more intense damping of the velocity than is the case after stalling at subsonic speeds. This is due to the flatter polars of the aircraft in the region of supercritical angles of attack at supersonic speeds (see Fig. 4.37). Following stalling with an initial force $n_{y0} > 1$, the beginning of the transitional section of spin finds the flight trajectory sloping upward (Fig. 4.55), due to the influence of a relatively high normal force directed in the initial moment (with no roll) vertically upward. This causes an increase in the irregularity of the aircraft's motion and complicates still further the pilot's task of orienting himself in this regime.

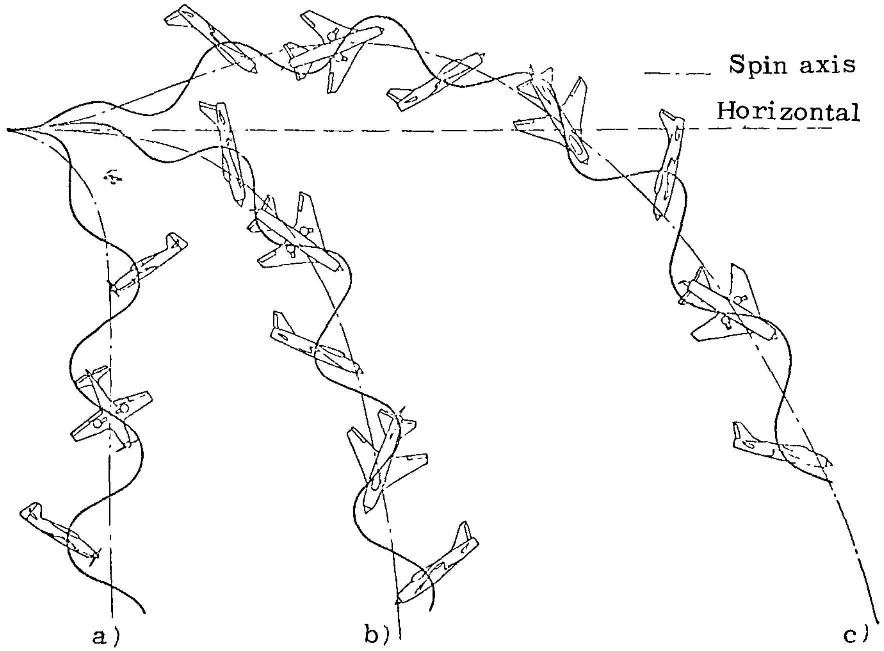

Fig. 4.55. Schematic Diagram of the Trajectory of an Aircraft in the Transitional Section of Spin:

(a) Subsonic Aircraft $V_0 = V_{min} (n_{y_0} = 1)$;

(b) Supersonic Aircraft: $V_0 = V_{min} (n_{y_0} < 1)$;

(c) Supersonic Aircraft: $V_0 > V_{min} (n_{y_0} > 1)$

CHAPTER 5

RECOVERY FROM SPIN

TERMINOLOGY AND RULE OF SIGNS

In considering the operation of control surfaces and ailerons by the pilot during spin and during recovery from it, we shall use the following generally employed terminology. When the elevator (controlled stabilizer) is tilted against the spin, it means that in normal spin the control stick (wheel) is pushed forward, while in inverted spin it is pulled back. The elevator is tilted with the spin when the control stick is pulled back in normal spin and pushed forward in inverted spin. When the rudder (moveable tail fin) is tilted with the spin, the left pedal is pushed forward in a left-hand spin (normal and inverted) and the right pedal is pushed forward in a right-hand spin. /183

When the rudder is deflected against the spin, the right pedal is pushed forward in a left-hand spin and the left hand pedal is pushed forward in a right-hand spin. Tilting the control surfaces for escape from spin means tilting them against the spin.

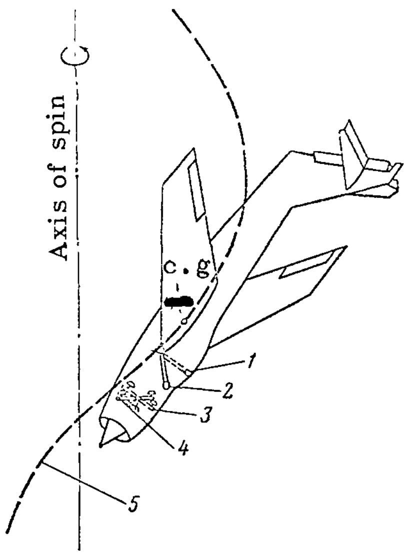

Fig. 5.1. Deflection Elevator and Rudder Against Spin in Right-Hand Inverted Spin (Ailerons in Neutral Position): (1) Control Stick Pulled Back (Against Spin); (2) Control Stick in Neutral Position; (3) Pedal in Neutral Position; (4) Left Pedal Pushed Forward (Against Spin; (5) Trajectory of Aircraft in Spin.

In Figs. 5.1 and 5.2, the positions of the control stick and control surfaces are shown for a right-hand inverted spin with the control surfaces tilted against the spin and with the spin.

When the ailerons are tilted with the spin, the right aileron is tilted upward in right-hand normal spin (the control stick is pushed toward the right side of the cockpit) or downward in right-hand inverted spin(the control stick is pushed toward the left side of the cockpit); in a left-hand spin, the situation is reversed.

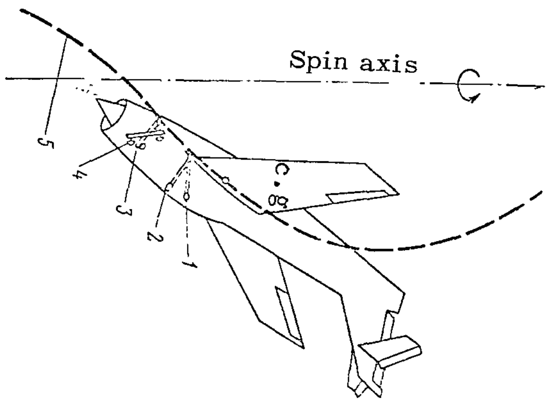

Fig. 5.2. Tilting of Elevator and Rudder with Spin in Right-Hand Inverted Spin (Ailerons in Neutral Position): (1) Control Stick in Neutral Position; (2) Control Stick Pushed Forward (With Spin); (3) Pedals in Neutral Position; (4) Right-Hand Pedal Pushed Forward (with Spin); (5) Trajectory of Aircraft in Spin. /18

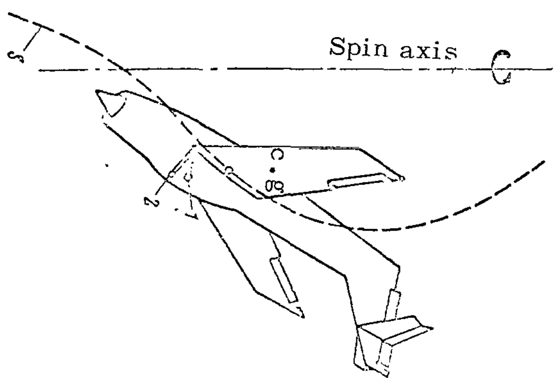

Fig. 5.3. Tilting of Ailerons With Spin in Right-Hand Inverted Spin (Elevator and Rudder Aligned With Spin); (1) Control Stick in Neutral Position; (2) Control Stick Pushed to the Left (With Spin); (3) Trajectory of Aircraft in Spin.

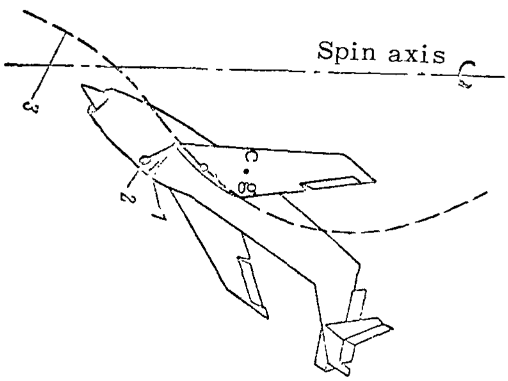

Fig. 5.4. Tilting of Ailerons Against Spin in Right-Hand Inverted Spin (Elevator and Rudder Aligned With Spin); (1) Control Stick in Neutral Position; (2) Control Stick Tilted to the Right (Against Spin); (3) Trajectory of Aircraft in Spin.

When the ailerons are tilted against spin, the right aileron is tilted downward in right-hand normal spin (with the control stick pushed to the left) or upward in right-hand inverted spin (control stick pushed to the right); in left-hand spins the situation is reversed. /185

Figs. 5.3 and 5.4 show the position of the control stick and ailerons, tilted with and against spin, in right-hand inverted spin.

The lag for recovery from spin is the time in seconds, or the number of turns made by the aircraft while spinning, between the moment when the tilting of the elevator has begun for escape from spin until the moment when the autorotation of the aircraft stops.

For the sake of clarity, the assumed rule of signs for the tilting of the control surfaces, ailerons and motion parameters of an aircraft in spin (averaged from the values of oscillating spin with sudden changes of sign in these parameters) can be represented in the form shown in Table 5.1.

TABLE 5.1

SIGNS FOR TILTING OF CONTROL SURFACES AND AILERONS, AND MOTION PARAMETERS OF AN AIRCRAFT IN A SPIN.

Spin Regime \ Parameter		Elevator (movable stabilizer)		Rudder (movable tail fin)		Ailerons		ω_x	ω_y	n_y
		with spin	against spin	with spin	against spin	with spin	against spin			
Normal Spin	left-hand	−	+	−	+	+	−	−	+	+
	right-hand	−	+	+	−	−	+	+	−	+
Inverted Spin	left-hand	+	−	−	+	−	+	+	+	−
	right-hand	+	−	+	−	+	−	−	−	−

5.1 Effect of Deflecting the Ailerons on Spin and Recovery From The Latter.

(a) PHYSICAL PICTURE OF THE EFFECT OF TILTING THE AILERONS

The characteristics of autorotation of an aircraft depend on the position of the ailerons during spin. When the ailerons are tilted during spin, the aerodynamic moments of roll $M_{x\text{a}} = M_x^{\delta}\text{a}\Delta\delta_\text{a}$ /186 and yaw $M_{y\text{a}} = M_y^{\delta_\text{a}} \Delta \delta_\text{a}$ are produced. The appearance of a moment relative to the longitudinal axis of the aircraft when the ailerons are tilted is caused by a change in the coefficients of normal aerodynamic force on the right and left wings, relative to its normal axis, and by the different amounts of change in the coefficients of tangential aerodynamic force on the right and left wings (tilting the ailerons up and down affects the tangential aerodynamic force differently). It has been found in experiments when the ailerons are tilted with spin that the moment $M_{x\text{a}}$ is directed toward ox_1. Tilting the ailerons against spin produces the moment $M_{x\text{a}}$ which resists rotation. The aerodynamic moment $M_{y\text{a}}$, which appears when the ailerons are tilted with spin, usually tends to slow down the rotation of the aircraft relative to its normal axis oy_1. When the ailerons are tilted against spin, the moment $M_{y\text{a}}$, directed towards increasing the rotation of the aircraft relative to the axis oy_1, appears in the majority of cases.

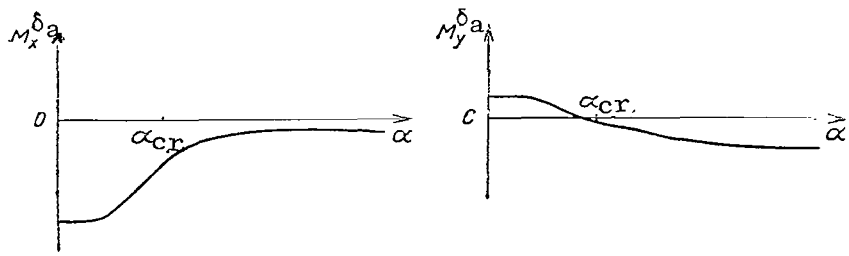

Fig. 5.5. Example of Curves for the Relationships $M_x^{\delta_\text{a}} = f(\alpha)$ and $M_y^{\delta_\text{a}} = \phi(\alpha)$ in a Supersonic Aircraft.

These considerations have been supported particularly by experimental data (Fig. 5.5) obtained for supersonic aircraft. The graph in Fig. 5.5 shows the curves for the derived aerodynamic moments (in a related system of coordinates) of roll $M_x^{\delta_\text{a}}$ and yaw $M_y^{\delta_\text{a}}$ in terms of the angle at which the ailerons are tilted as a function of the angle of attack of the aircraft. It is clear from the graph that the derivative $M_x^{\delta_\text{a}}$ does not change sign even with transition to supercritical angles of attack, while the derivative $M_y^{\delta_\text{a}}$ changes its sign even at near-critical angles of attack. /187

Hence, if the aircraft in question is in a left-hand normal spin, for example, the following picture develops. When the ailerons are tilted with spin ($\Delta\delta_\text{a} > 0$) the moment of roll $M_{x\text{a}} = M_x^{\delta_\text{a}} \Delta\delta_\text{a} < 0$ (since $M_x^{\delta_\text{a}} < 0$) and yaw $M_{y\text{a}} = M_y^{\delta_\text{a}} \Delta\delta_\text{a} < 0$ ($M_y^{\delta_\text{a}} < 0$ in spin) are produced, which tend to roll the aircraft onto its left wing and to push the nose of the aircraft to the right. In other words, the moment of roll tends to increase the rotation of the aircraft in a left-hand spin and the moment of yaw impedes the rotation of the aircraft to the left.

In the general case, the moments $M_{x\text{a}}$ and $M_{y\text{a}}$ in combination can have different effects on spin in different types of aircraft, (and consequently, on the escape of such aircraft from spin). This is explained by the fact that when the ailerons are tilted, the aerodynamic moments of the wing are changed; since they depend on the magnitude and direction of the other moments acting on the aircraft during the spin, they can have different effects on the spin regime and the characteristics of escape from it. in the majority of cases, tilting the ailerons during spin and during escape from it has a very significant effect on the nature of the regime, which can be explained primarily by changes in sideslip under the influence of the tilted ailerons.

(b) TILTING OF THE AILERONS IN NORMAL SPIN

The effect of tilting the ailerons to produce spin in supersonic aircraft is much more pronounced in comparison to subsonic aircraft. In supersonic aircraft, which have relatively short wings, tilting the ailerons with spin usually makes the regime of normal spin less stable, reduces the absolute values of the average angular velocities, and leads to an increase in the oscillations of the aircraft (especially the longitudinal and transverse ones), increases the irregularity of rotation, and can even cause a spontaneous change in the direction of the aircraft's rotation. This may cause periodic interruptions in the movement of the aircraft in a regime, causing it to roll in the direction opposite to the direction of rotation, etc. Tilting the ailerons against spin in a regime of normal spin usually causes a more stable and uniform spin with smaller oscillations of the aircraft, which in some cases occur at greater average angles of attack; occasionally, tilting the ailerons agsinst spin causes the aircraft to shift from normal to inverted spin. In certain cases, tilting the ailerons against spin has a very small or practically negligible influence on the regime.

Let us examine in greater detail the influence of tilting the ailerons in normal spin, upon the characteristics of the latter by using a number of examples obtained in the course of flight tests of spin in supersonic aircraft. Let us begin with spins which were produced following stalling at initial altitudes on the order of H_0 = 10-12 km. In Figs. 5.6 and 5.7 we have plotted, for the sake of comparison, the characteristics of left-hand normal spin with ailerons in a neutral position in the regime and with the ailerons tilted with spin. It is apparent from Fig. 5.7 that the pilot tilted the ailerons with spin immediately following stalling, to an angle of $\Delta\delta_a \approx 15$-17°, and then held them in this position until the aircraft began to emerge from the spin. About 10 sec after the spin began, ($t \approx 16$ sec in Fig. 5.7) the rotation of the aircraft stopped, after which it spontaneously shifted from left-hand to right-hand normal spin, which continued with the control stick tilted for left-hand spin (i.e., against right-hand spin). The effect of tilting the ailerons against the spin, expressed in the characteristics of left-hand normal spin, are shown in Fig. 5.8. In this case, the ailerons were tilted to $\Delta\delta_a \approx 5$-8°, which produced a certain increase in the average value of the angle of velocity of yaw.

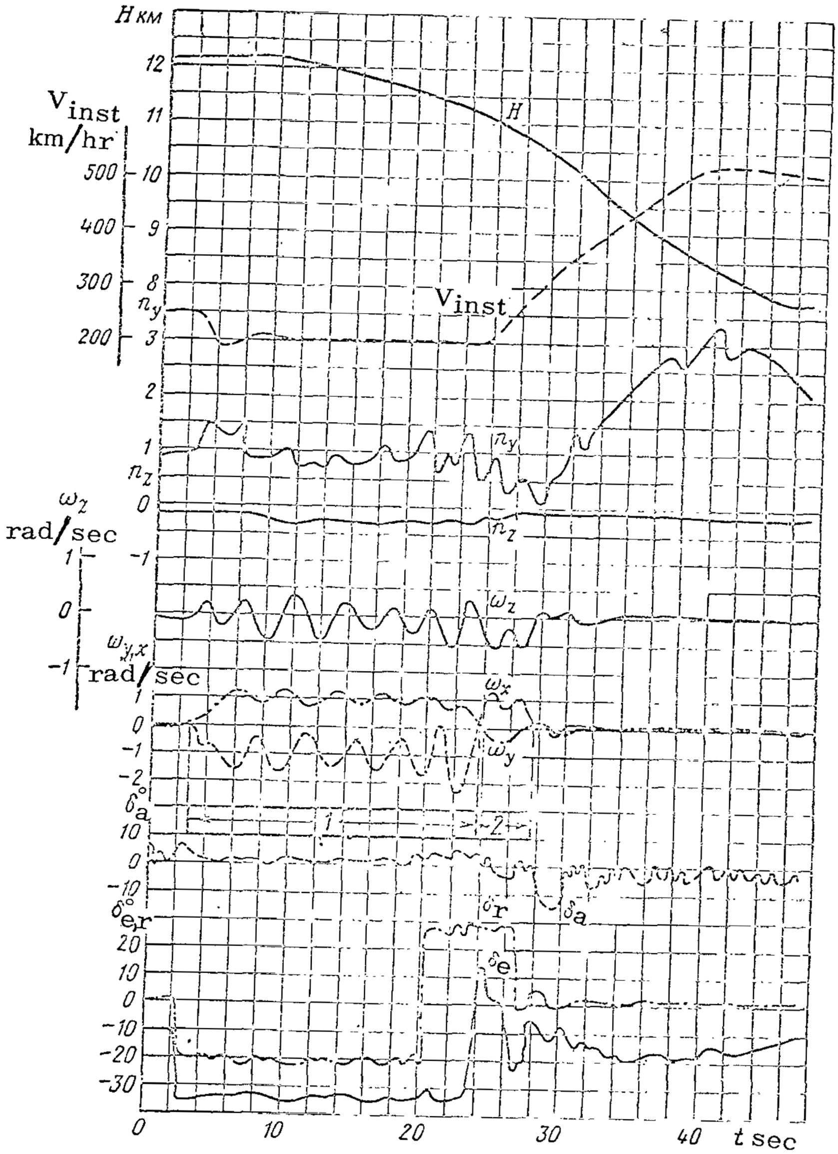

Fig. 5.6. Left-Hand Normal Spin (Initial Regime) with Ailerons in Neutral Position: (1) Left-Hand Normal Spin; (2) Right-Hand Normal Spin.

The effect of tilting the ailerons with spin in right-hand normal spin to an angle of $\Delta\delta a \approx 12°$ is shown in Fig. 5.9. It is evident from the graph that during the first seven seconds or so after the ailerons were tilted, there was a highly irregular regime of right-hand spin, after which the aircraft spontaneously entered a left-hand normal spin. The latter was more stable and uniform, and involved fewer oscillations than did the left-hand normal spin of the same aircraft with neutral ailerons (see Fig. 5.6). Fig. 5.10 shows right-hand normal spin during which the pilot tilted the ailerons with spin to an angle of $\Delta\delta a \approx 8$-$11°$. This tilting of the ailerons did not have a significant effect on the nature of the regime, as in the preceding case. The spin remained right-hand normal, but proceded with greater oscillation of the aircraft during the regime. Tilting the ailerons against the spin to an angle $\Delta\delta_a \approx 12°$ in right-hand normal spin is shown in Fig. 5.11. The longitudinal oscillations of the aircraft increased when this was done. /189

(c) TILTING THE AILERONS IN INVERTED SPIN.

The effect of tilting the ailerons can also be very important in inverted spin. Tilting the ailerons with spin in supersonic aircraft in a regime of inverted spin (as well as in normal spin) usually produces an increase in the irregularity of the motion and oscillation of the aircraft reducing the stability of the motion in this regime and possibly leading to interruptions or changes in the rotation direction of the aircraft and sometimes even to a shift of the aircraft from normal to inverted spin. Tilting the ailerons against the spin in a regime of inverted spin produces a reduction of the aircraft oscillation, occurrence of more uniform rotation, and a transition to average angles of attack with a smaller absolute value, i.e., the nose of the aircraft drops while it is on its back, which sometimes leads the aircraft to enter a regime of normal spin.

Fig. 5.12 shows an example illustrating the influence of tilting the ailerons with spin, on right-hand inverted spin. Tilting the ailerons in this regime was generally done to give a value of $\Delta\delta_a \approx 20°$. It is clear from the graph that tilting the ailerons in this manner produced a non-uniform unstable spin. Tilting the ailerons with spin to an angle of $\Delta\delta_a \approx 8°$ in the process of pulling the airplane out of its position on its back into a left-hand inverted spin is shown in Fig. 5.13. It is clear from the example given that tilting the ailerons in this manner causes the aircraft to shift to left-hand normal spin. This normal spin took place with the control surfaces tilted for inverted spin. Fig. 5.15 shows an example in which the ailerons were tilted against spin to an angle $\Delta\delta_a \approx 14$-$18°$ in a regime of left-hand inverted spin. In this case, the tilting of the ailerons caused the aircraft to drop its nose (the negative angle of attack decreased), so that the aircraft began to rotate in a practically vertical position at a high angular velocity of roll ($\Delta t \approx 27$-33 sec, $\omega_{x\,av} \approx 2.8$ rad/sec, $|\omega_{y}av| \approx 0.1$ rad/sec).

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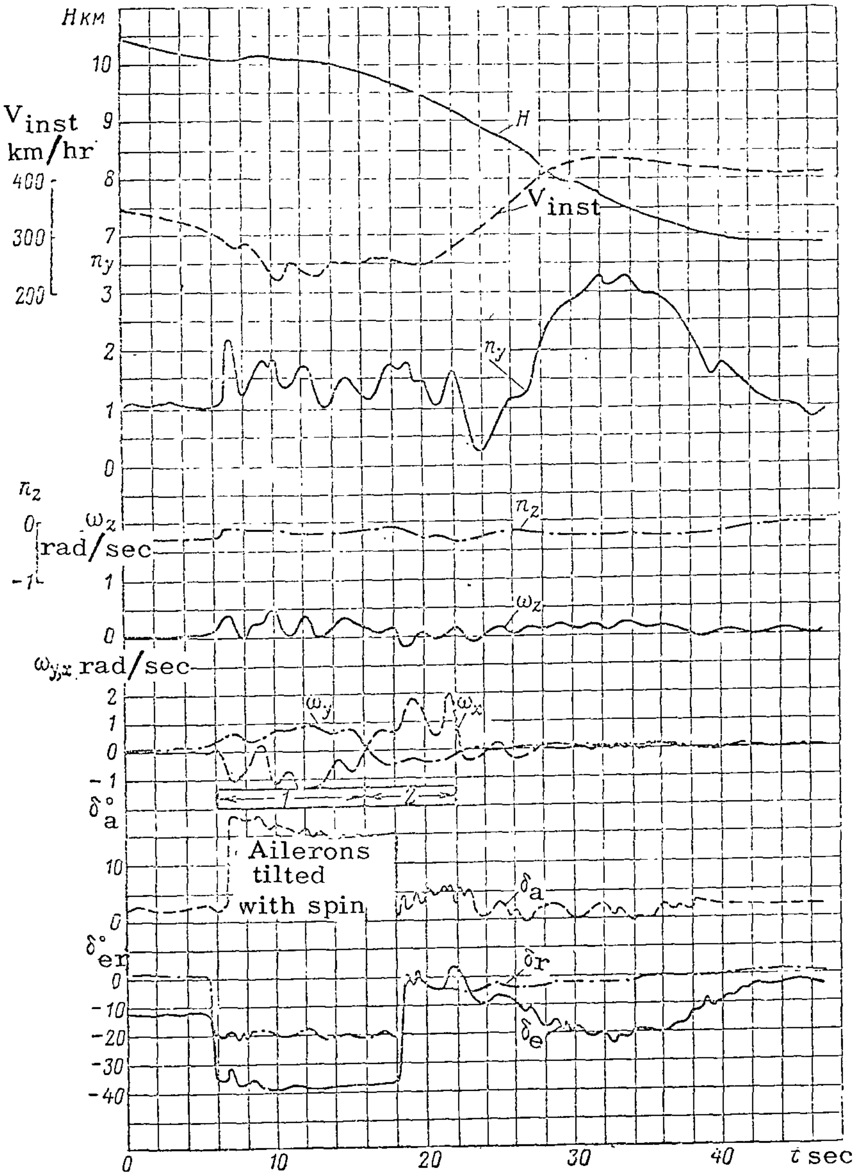

Fig. 5.7. Left-Hand Normal Spin (Initial Regime) with Ailerons Tilted With Spin: (1) Left-Hand Normal Spin (2) Right-Hand Normal Spin.

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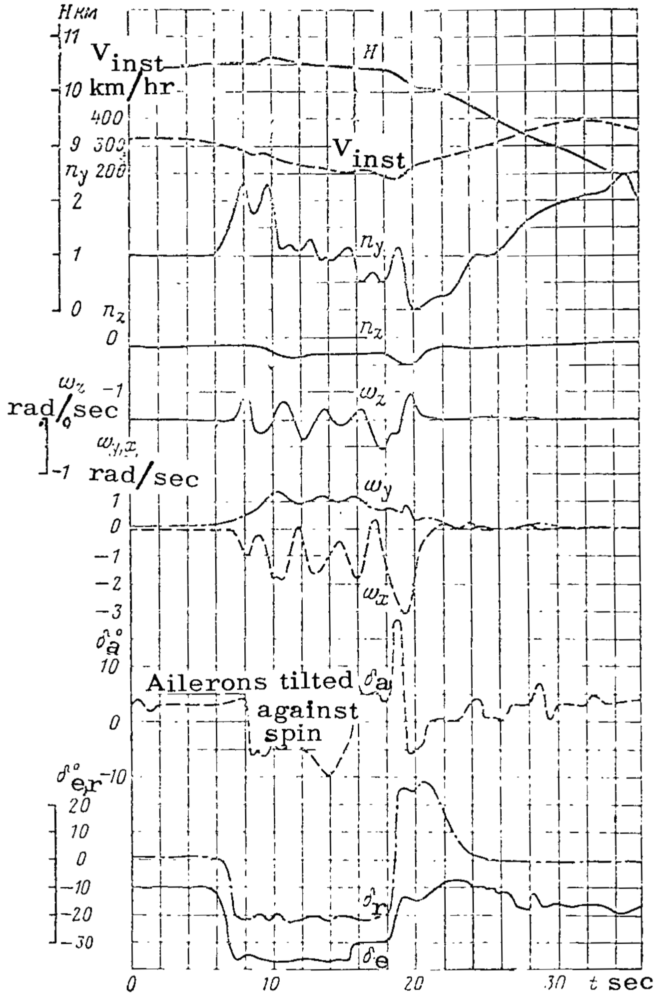

Fig. 5.8. Left-Hand Normal Spin ($\Delta t \approx$ 7-21 sec) with Ailerons Tilted Against Spin ($\Delta t \approx$ 8-15.5 sec).

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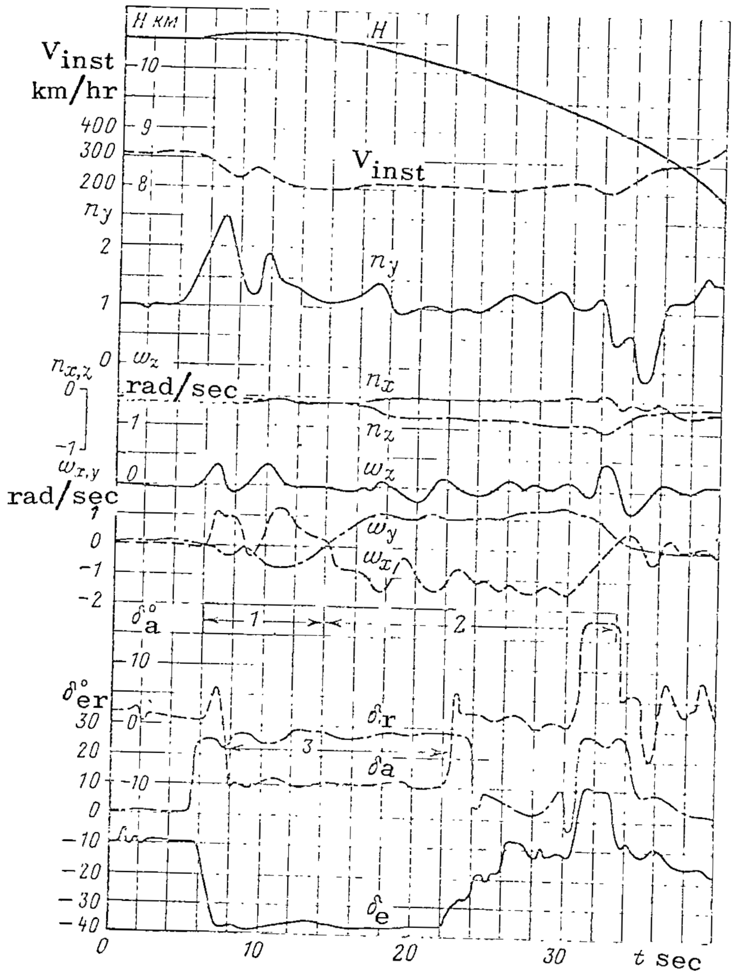

Fig. 5.9. Tilting of Ailerons ($\Delta t \approx$ 7-25 sec) with Spin to a Value of $\Delta\delta_a \approx 12^\circ$ in Right-Hand Normal Spin (Initial Regime) (1) Right-Hand Normal Spin; (2) Left-Hand Normal Spin; (3) Ailerons Tilted with Spin.

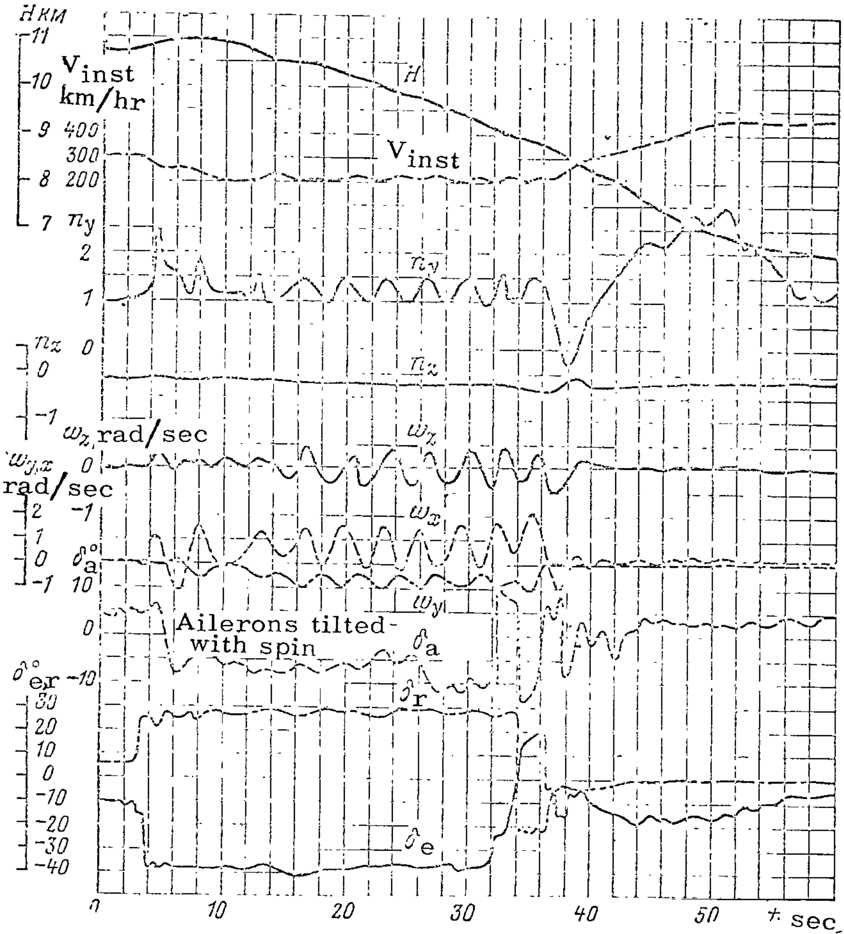

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Fig. 5.10 Tilting of Ailerons ($\Delta t \approx$ 5-32 sec (with Spin to a Value of $\Delta\delta_a \approx$ 8-11° in Right-Hand Normal Spin.

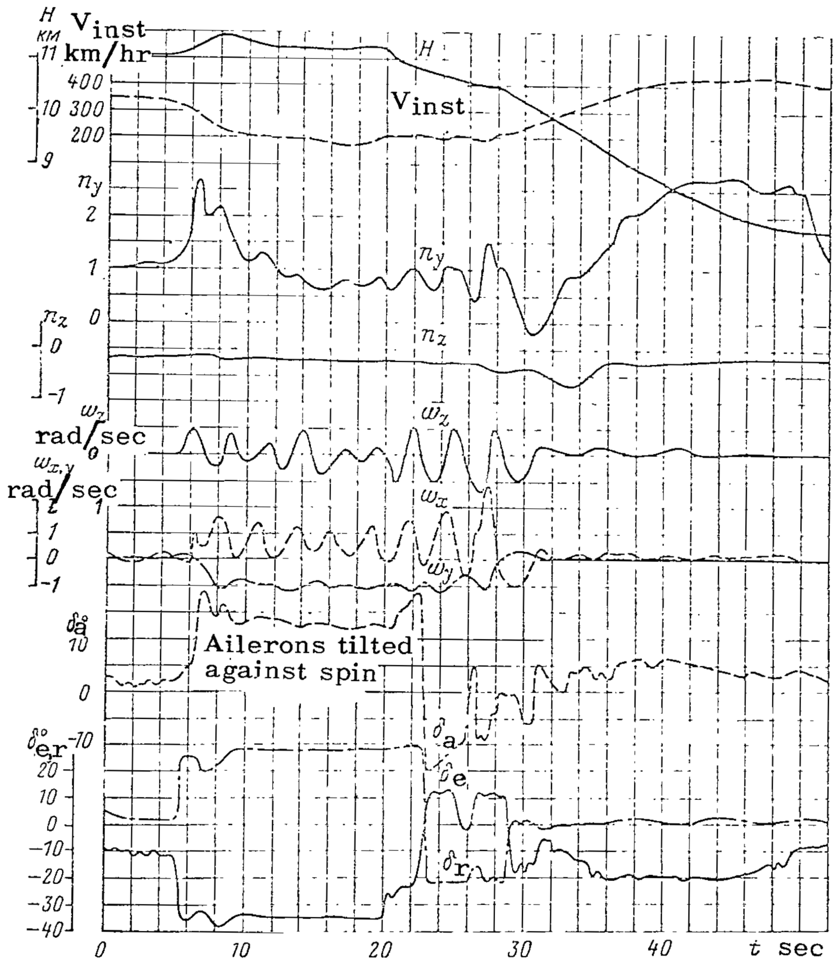

Fig. 5.11 Tilting of Ailerons ($\Delta t \approx 7$-23 sec) Against Spin to an Angle of $\Delta\delta_a \approx 12°$ in Right-Hand Normal Spin.

The experimental (flight) data obtained support the fact that in both normal and inverted flight, the effect of tilting the ailerons on the characteristics of a regime always depends to a considerable degree on the magnitude of the angle to which they are tilted and has practically no relation to the elapsed time following the beginning of the spin regime until they were tilted. Tilting the ailerons has practically the same effect on the characteristics of right- and left-hand spins. /195

(d) INFLUENCE OF TILTING THE AILERONS AT HIGH ALTITUDES.

The effect of tilting the ailerons on spin has a very definite relationship to the flight altitude. At high altitudes, tilting the ailerons has the same qualitative effect as at low altitudes, but the effect usually is more pronounced. Tilting the ailerons with spin in a regime of normal spin produces greater oscillations of the aircraft and increased irregularity of rotation, increases the tendency of the aircraft to change its direction of rotation, etc. Sometimes tilting the ailerons during spin at high altitudes causes the aircraft to "follow its ailerons", (i.e., their effect is direct, as in other operational regimes of flight), which increases still further the irregularity of the regime and can even aggravate the tendency for the aircraft to shift to spin in the other direction or even to enter inverted spin. The latter is explained by the aircraft's entering subcritical angles of attack during such spin, thus producing a direct effect on the ailerons (which is usual in normal operational flight regimes).

Let us examine several examples which illustrate the effect of tilting the ailerons with spin after stalling at high altitudes. Fig. 5.15 shows an example of tilting the ailerons with spin in a regime of left-hand normal spin, beginning at an altitude of $H_0 \approx 18.5$ km. During the first moment following the tilting of the ailerons ($t \approx 19$ sec in the graph), the aircraft "travelled behind the ailerons". A highly unstable left-hand spin then began, which caused the aircraft to flutter downward like a leaf along a spiral trajectory, with considerable oscillation. Beginning at $t \approx 30$ sec, the aircraft spontaneously shifted to a right-hand normal spin, involving still greater longitudinal and transverse oscillations of the aircraft. Setting the ailerons to the initial balanced position ($t \approx 49$ sec) caused the aircraft to shift once again to left-hand normal spin, which also involved fluttering of the aircraft. Tilting the ailerons against spin in a left-hand normal spin occuring at an initial altitude of $H_0 \approx 18.5$ km is illustrated in Figs. 5.17 and 5.18. It is clear from Fig. 5.17 that tilting the ailerons with spin in this particular case led to a highly unstable spin which took the form of flutter downward along a spiral trajectory, with considerable oscillation of the aircraft. The angles of roll of the aircraft reached 180°, i.e., the aircraft appeared to be flying on its back. Tilting the ailerons against spin in the example shown in Fig. 5.18 causes the aircraft first to "follow its ailerons" ($t \approx 35$ sec), then to drop its nose and go over onto its back; it

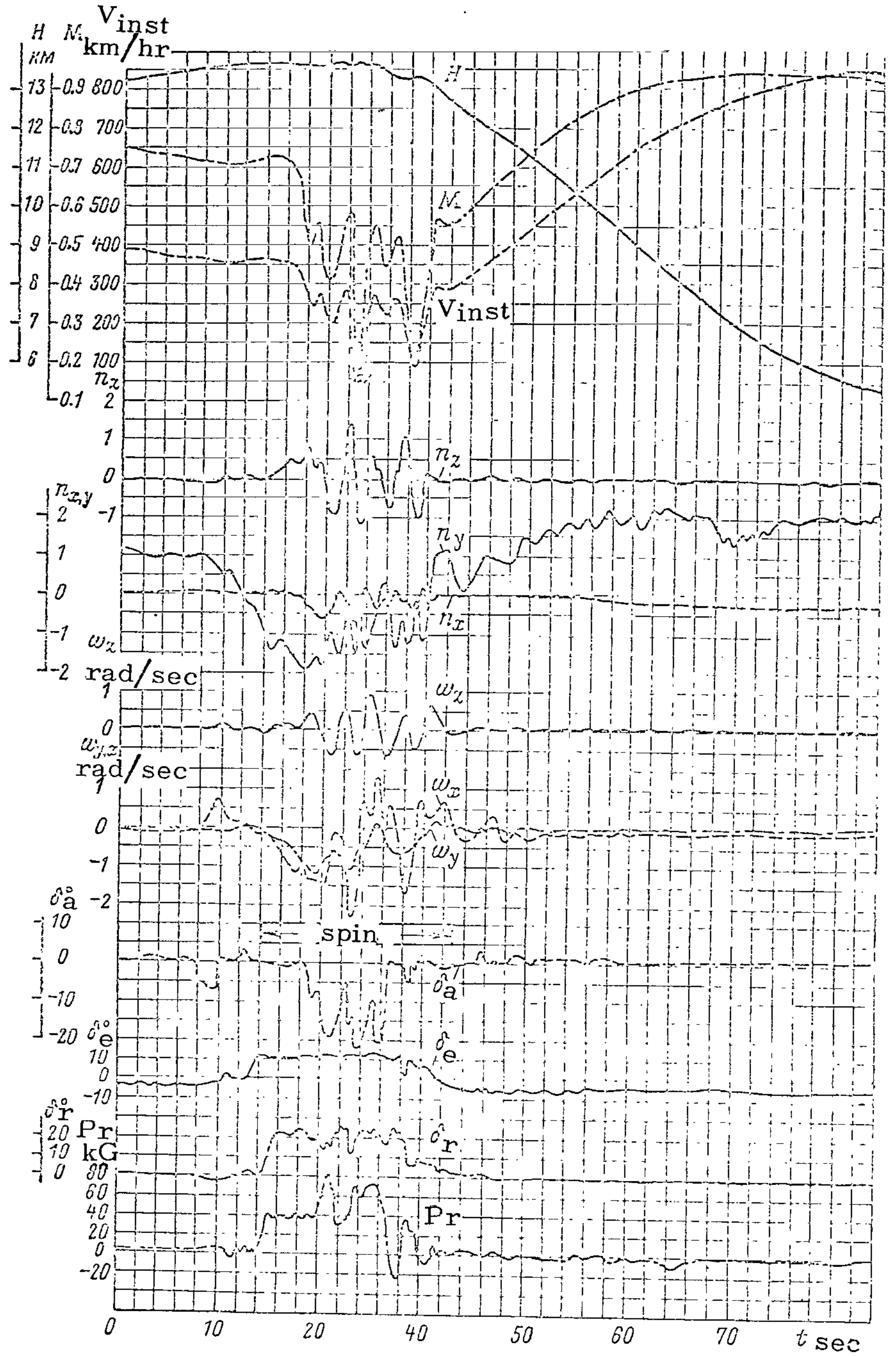

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Fig. 5.12. Tilting of Ailerons ($\Delta t \approx$ 19-26.5 sec) with Spin to an Angle $\Delta\delta_a \approx 20°$ in Right-Hand Inverted Spin.

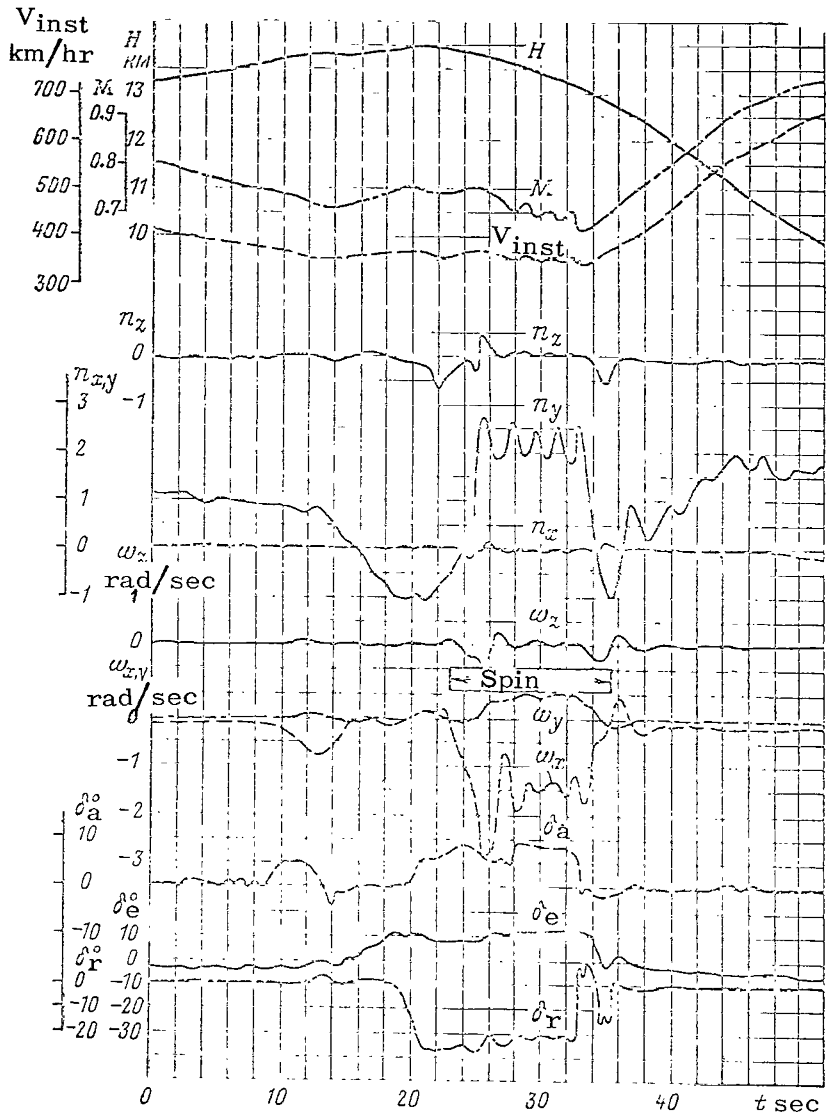

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Fig. 5.13 Tilting of Ailerons ($\Delta t \approx$ 20-33 sec) with Spin to an Angle $\Delta\delta_a \approx$ 8° with an Attempt to Place the Aircraft in a Left-Hand Inverted Spin.

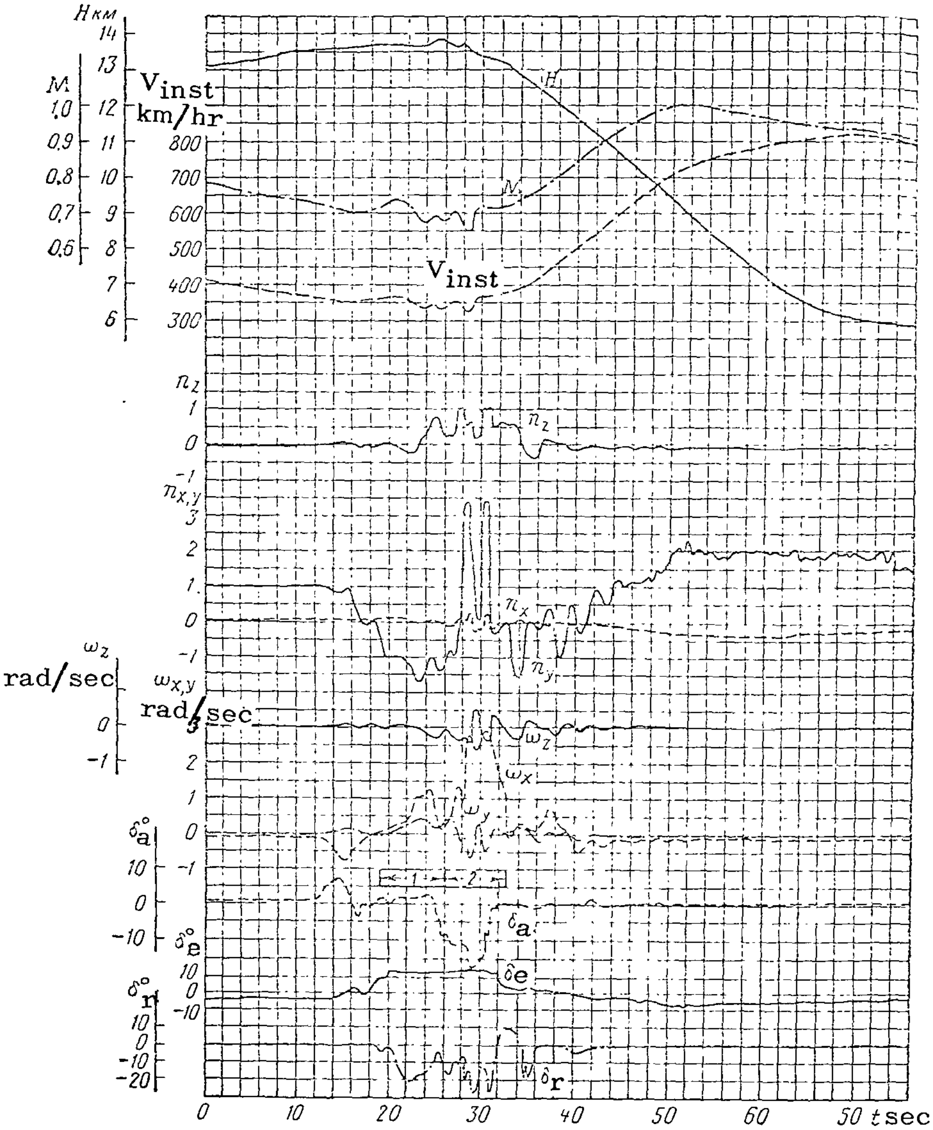

Fig. 5.14. Tilting of Ailerons ($\Delta t \approx$ 25-31 sec) Against Spin to an Angle $\Delta\delta_a \approx 14° - 18°$ in Left-Hand Inverted Spin: (1) Left-Hand Inverted Spin; (2) Right-Hand Normal Spin; (3) Ailerons Tilted Against Spin.

entered an inverted spin with the control surfaces tilted for normal spin. Inverted spin lasted for about 4 sec (t 38-42 sec). Setting the ailerons and rudder to a neutral position caused the aircraft to return to a right-hand normal spin. The rudder was again tilted with the spin while the ailerons were tilted against the spin. It is clear from the graph that this caused a repetition of the phenomenon described above: the aircraft again shifted from normal to inverted spin ($t \approx 52$ sec). /199

In aircraft without any boosters in the control system, tilting the ailerons in spin can lead to a considerable change in the force applied to the pedals (effect of changing sideslip, taking the form of vibration with considerable and abrupt periodic changes in the hinge moments of the rudder, caused by changes in the airflow over the vertical tail. A spin of this kind is shown in Fig. 5.19. It is clear from the graph that the abrupt shifting of force on the pedals (ΔP_r >60 kG) did not give the pilot an opportunity to hold the pedals in the fixed position required for this regime, which is all the way over with spin.

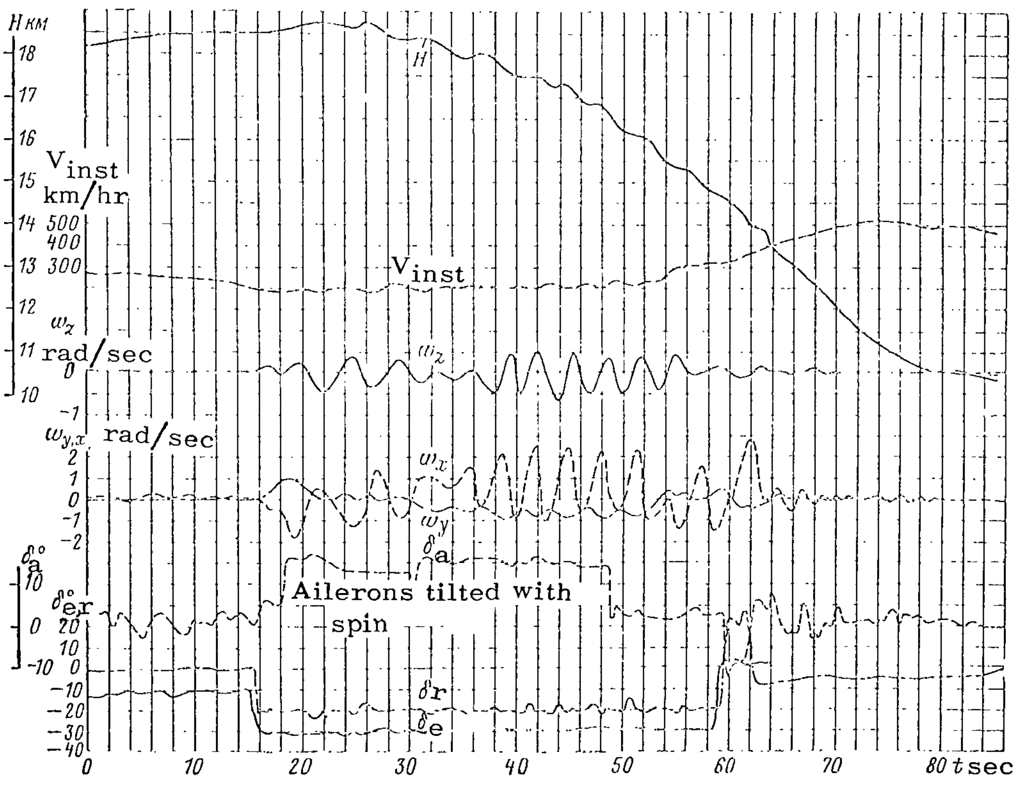

Fig. 5.15. Left-hand Normal Spin Beginning at Initial Altitude $H_0 \approx 18.5$ km with Ailerons Tilted ($\Delta t \approx 19$-49 sec) with Spin. /200

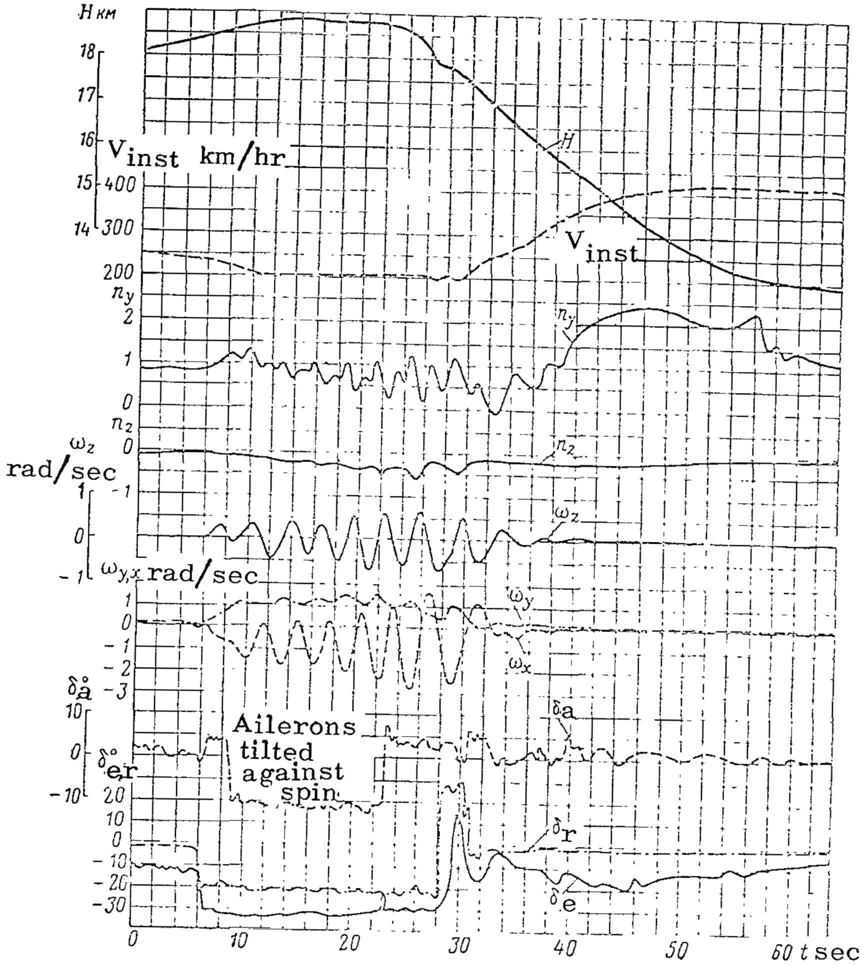

Fig. 5.16. Left-Hand Normal Spin ($\Delta t \approx$ 9-30 sec) Beginning at Initial Altitude $H_0 \approx$ 18.5 km with Ailerons Tilted Against Spin ($\Delta t \approx$ 9-23 sec). /201

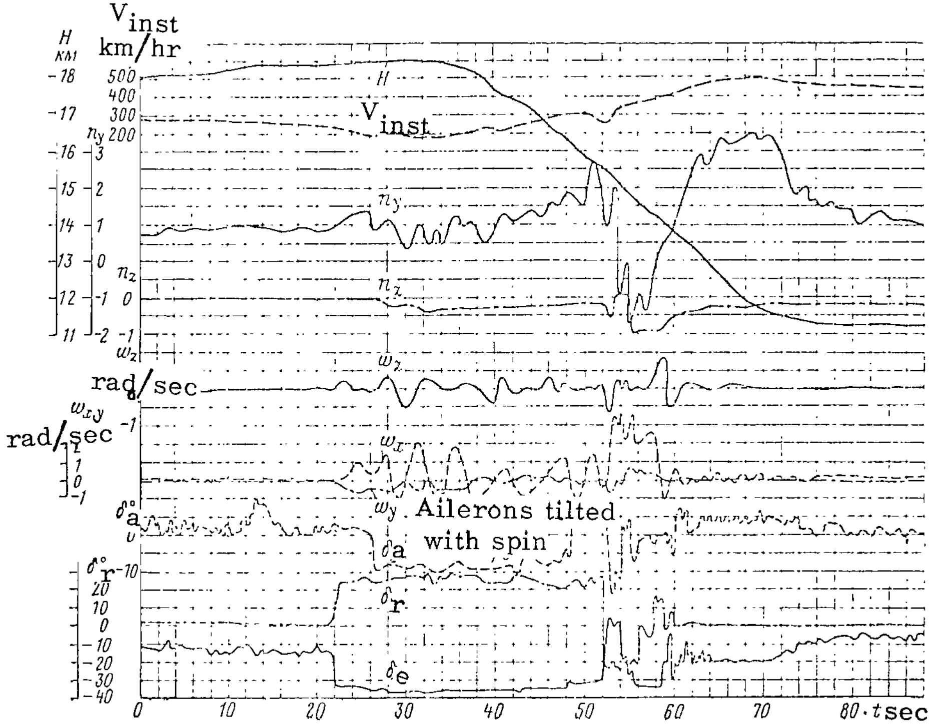

Fig. 5.17. Right-Hand Normal Spin Beginning at Initial Altitude $H_0 \approx 18.5$ km with Ailerons Tilted with Spin ($\Delta t \approx 26\text{-}48.5$ sec).

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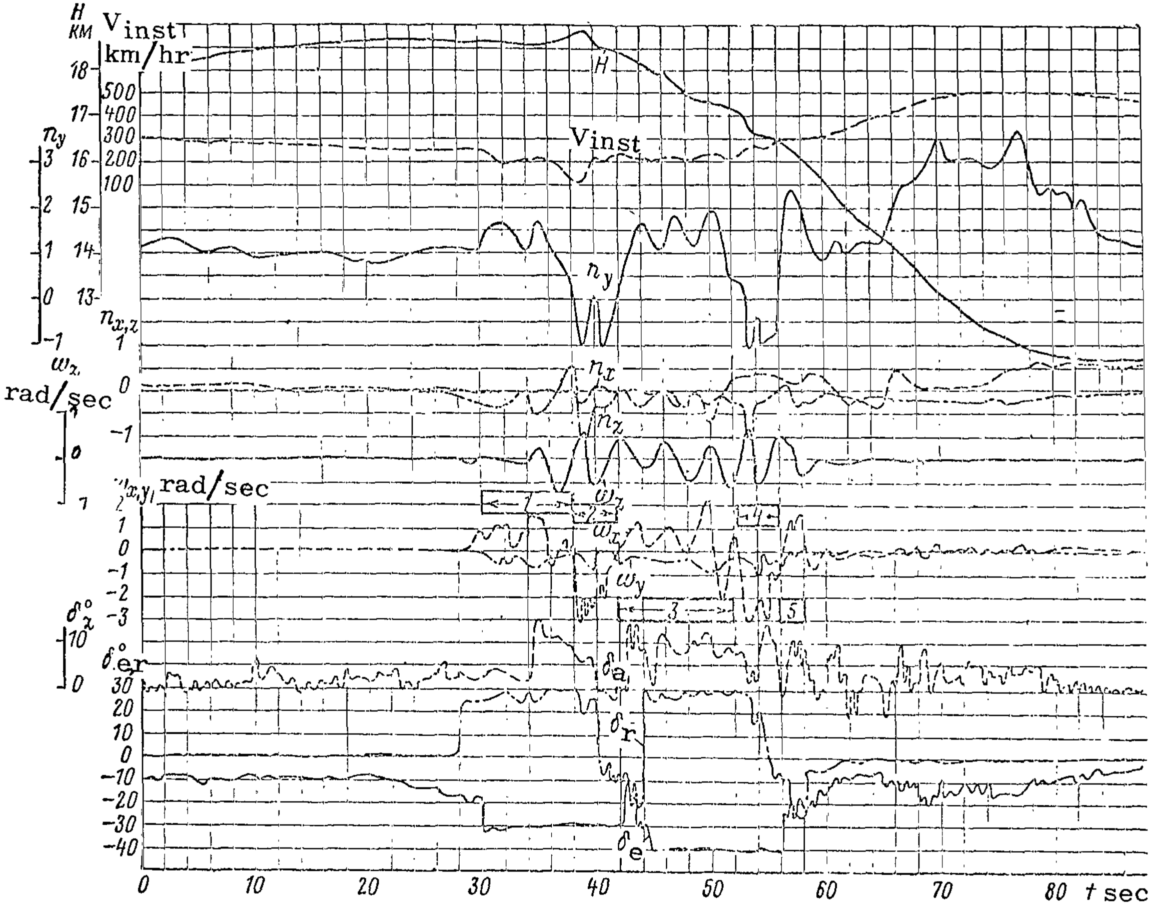

Fig. 5.18. Right-Hand Normal Spin (Initial Regime) Beginning at Initial Altitude $H_0 \approx 18.5$ km, with Ailerons Tilted Against Spin) ($\Delta t \approx 34.5$-40 sec and 43-58 sec) (1) right-hand normal spin; (2) right-hand inverted spin; (3) right-hand normal spin; (4) right-hand inverted spin; (5) right-hand normal spin.

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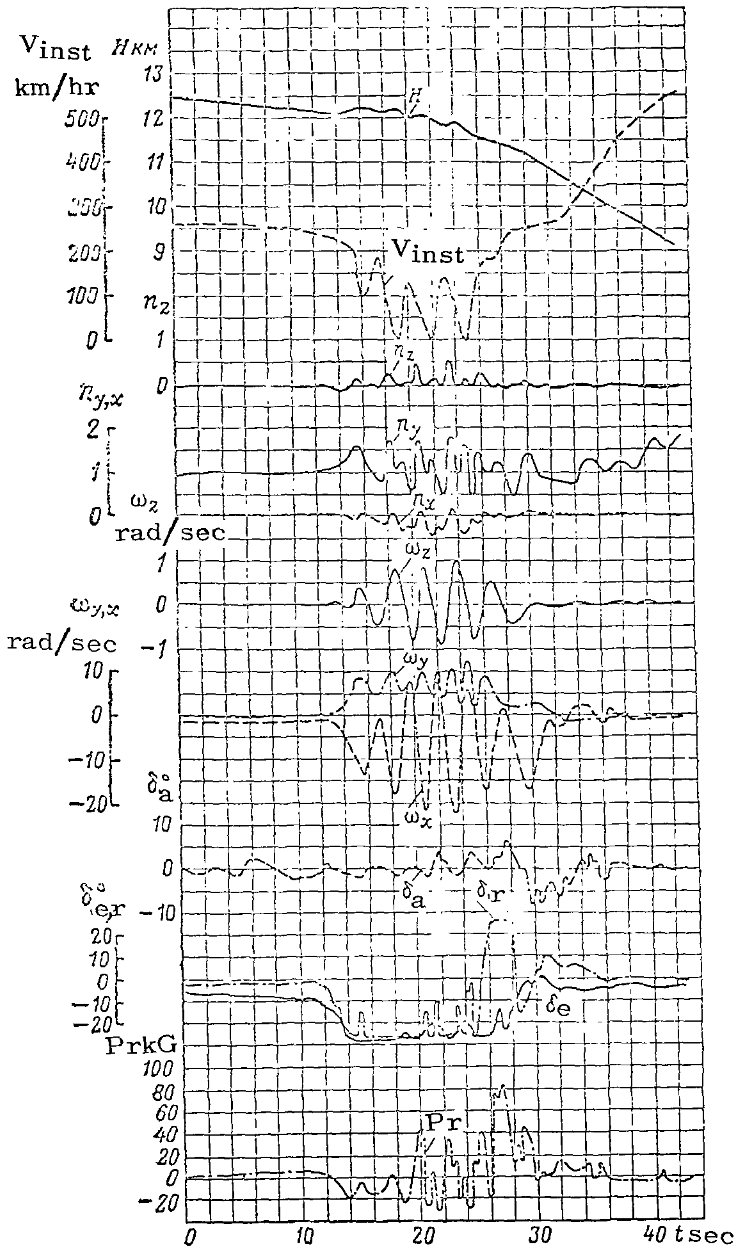

Fig. 5.19. Left-Hand Normal Apin ($\Delta t \approx$ 15-31 sec) During Which Tilting of the Ailerons Produced Sharp Reduction in the Forces on the Pedals.

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It is clear from what has been stated above that the effect of the ailerons in the spinning of supersonic aircraft can be highly diverse and is usually quite pronounced. In view of such diversity and intensity of the effect produced by tilting the ailerons during spin, the pilot must always try to keep the ailerons in the initial balance position, if possible, both during the regime and when escaping it (except in Escape Method Number 4, which will be discussed below). This is especially true at high altitudes. At low subsonic flight speeds, this is usually very close to a neutral position. /2

5.2. Methods of Pulling a Supersonic Aircraft Out of a Spin.

(a) GENERAL CHARACTERISTICS OF METHODS OF RECOVERY

Maneuvering an aircraft in a spin is very complex and quite different from maneuvering it in all other flight regimes. To pull an aircraft out of a spin, it is necessary to use special methods of pilotage. These methods are basically different from the methods of pilotage used in all other operational regimes of flight. This is due primarily to the fact that the reaction of the aircraft to the motion of the control surfaces in a spin occurs much more slowly, while the motion of the control surfaces must be sharper and more pronounced; frequently, they must be moved all the way from stop to stop. The existence of high angular velocities produces significant changes in all the fundamental characteristics of maneuverability of an aircraft (see Chap. II). In supersonic aircraft as a rule, the action of the ailerons in a spin is opposite to that which takes place under usual flight regimes, while tilting the ailerons frequently has a deciding influence on the nature of the spin. In fact, the actual methods of terminating a spin, are to a certain degree, directly opposite to the customary natural methods used in all other flight regimes. Thus for example, if we consider merely the position of the aircraft in space, we can draw a convenient formal analogy between spin and dive: in both cases, the position of the aircraft in space is the same (its nose is down). However, in order to pull an aircraft out of a dive, it is necessary to pull the control stick back; to escape from a spin, on the other hand, the control stick must be pushed forward, since the physical principles involved in these two cases are basically different. We should add in this regard that in a spin, which is a very unusual (and fortunately relatively rare) flight regime, the sensations and possibility for spatial orientation by the pilot are fundamentally different from those in conventional flight regimes (the effect of the angular velocities and forces, the unusual and rapidly changing position of the aircraft in space, the lack of visibility or only partial visibility of a /20 portion of the horizon, etc.). The considerable complication of operating conditions and orientation problems of the pilot in a spin makes demands on his presence of mind, promptness of action, attention, skill, as well as considerable demands on his physical powers to carry out the precise and accurate movements of the controls which are required to pull the aircraft out of a spin.

Pulling an aircraft out of a spin consists essentially of three stages (see Fig. 4.32):

(a) The first and basic stage is simply escape from spin, i.e., termination of the autorotation of the aircraft;

(b) The second stage is a dive (even at subcritical angles of attack), to increase the flight speeds in order to make it possible to carry out further maneuvers with the aircraft safely;

(c) The third stage consists in pulling the aircraft out of the dive and placing it in a regime of straight-line horizontal flight at a speed equal to or greater than the original.

We usually understand the term "method of escape from spin" to be a method of pilotage which insures a reliable termination of autorotation of the aircraft, and is only the first step in pulling the aircraft out of the spin. If we retain the generally adopted terminology, we can discuss the methods of piloting the aircraft in the first stage of escape from spin in this particular section and refer to them as "escape methods". We will discuss the remaining two stages of escape from spin later on. From now on we shall be using two terms which refer to the same phenomenon: "pulling out of" and "escaping from" a spin. The first of these will be used when referring to the actions of the pilot (he pulls the aircraft out of a spin); the second will be applied when describing the motion of the aircraft itself under these conditions (the aircraft escapes from a spin). To a certain degree, these two terms are interchangeable.

The first and most important stage in pulling out of a spin is considered complete when the average angle of attack of the aircraft is subcritical, causing the autorotation of the aircraft to stop.

(b) PULLING OUT OF A NORMAL SPIN

Four methods are used for pulling modern aircraft out of normal spin (stopping autorotation); they are shown schematically in Fig. 5.20.

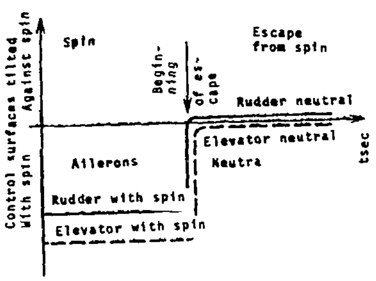
(1) Method 1: Pulling out of a spin by simultaneously setting the elevator (movable stabilizer) and rudder in a neutral position, with the ailerons in a neutral position.

(2) Method 2: Pulling out of a spin by tilting the rudder all the way against the spin and then (after 2-4 sec) setting the elevator in a neutral position, with the ailerons in a neutral position.

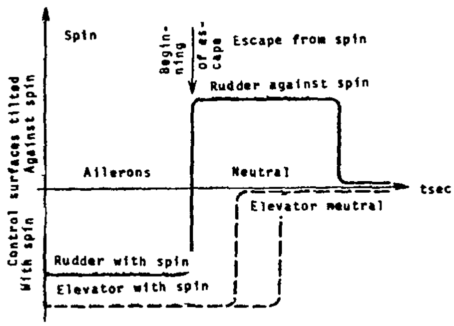
(3) Method 3: Pulling out of a spin by tilting the rudder all the way against the spin, then tilting the elevator all the way against the spin, with the ailerons in a neutral position. /207

(4) Method 4: Pulling out of a spin in the same way as in Method 3, but tilting the ailerons as far as possible at the same time that the rudder is tilted; in supersonic aircraft, as a rule, this amounts to tilting them with the spin.

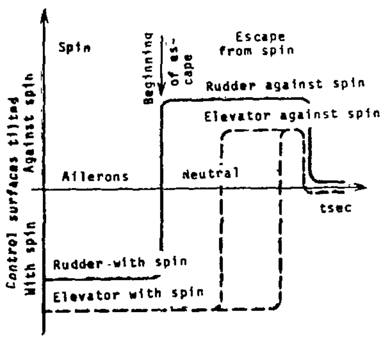
Method No. 1



Method No. 2



Method No. 3



Method No. 4

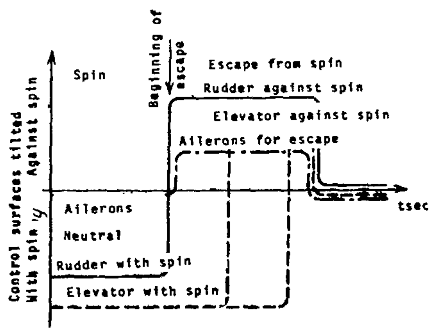


Fig. 5.20. Schematic Representation of the Movements of the Control Surfaces in the Different Methods of Pulling an Aircraft Out of a Normal Spin.

The methods of pulling an aircraft out of a normal spin, as we can see from the above, are listed in the order of increasing "strength" i.e., effectiveness (the values of the aerodynamic moments created by the control surfaces which serve to pull the aircraft out of the spin). The "strongest" (and most effective) method is Number 4.

Method 1 is recommended for pulling an aircraft out of an unstable normal spin; Method 2 is for unstable oscillating spin; Method 3 is for stable uniform spin, and Method 4 is for highly stable uniform normal spin, occuring at very high average angles of attack (see Fig. 4.18). The use of these methods makes it possible to pull modern aircraft out of all possible regimes of normal spin more rapidly and reliably (with a minimum loss of time and altitude in /208 the process). Examples of pulling supersonic aircraft out of a spin, using the four methods described, are shown in Fig. 4.22 (Method 1), Fig. 4.26 (Method 2, Fig. 4.28 (Method 3), and Fig. 4.29 (Method 4); escape began at $t \approx 66.5$ sec.

When an aircraft enters a spin, it is always necessary to adjust the control surfaces completely with the spin. This is because the maximum effect produced by the action of the control surfaces when pulling the aircraft out of a spin is obtained by moving the surfaces out of the position with spin. In this case, it is possible to obtain a maximum excursion for the movable surfaces, from one stop to the other. In addition, it is also possible to employ the dynamic ("shock") effect in moving the control surfaces abruptly from one extreme position to the other. This maximum exploitation of all possibilities in order to escape spin is only required when the aircraft falls into a sufficiently stable spin regime. However, since the pilot does not know in advance which spin regime is developing, he must always put the control surfaces all the way over in the direction of spin immediately.

The fact that there are now four methods which have been devised in the course of special flight tests on modern aircraft for developing ways of escaping from spin, instead of the one (the so-called standard method, Method 3) which was formerly recommended, considerably increases the chances of pulling a modern aircraft out of a spin, and also increases the safety of flight, although it places some additional demands on the pilot's attention. The experience of flight tests and considerable studies of supersonic aircraft, as well as the statements of many highly qualified pilots and those who are less qualified indicate that this difficulty is much less than would appear at first glance, for the following reason. The four methods which we have described for pulling out of a spin differ only in a gradual "increase" (an increase in the motion required for the control surfaces and the intervals between their motions), since the operating principle for the surfaces remains essentially the same (the ailerons need be moved only when using Method 4).

It is not possible to limit ourselves to a use of only the "strongest" methods (Nos. 3 and 4) as is sometimes erroneously suggested by those who wish to simplify the work of the pilot. This is not feasible for the following reasons: in supersonic aircraft, particular attention is usually paid when discussing these problems to the provision of sufficient effectiveness of the control surfaces in order to pull the aircraft out of a spin. This is because when a subsonic aircraft falls into a spin, there is a considerable danger that the pilot may not be able to pull out, due to the low effectiveness of the control surfaces in pulling out of a spin, (danger of a "shortage" of control surfaces when trying to pull out of a spin).

In supersonic aircraft, the required effectiveness of the control surfaces comes primarily from the flight conditions at high Mach /2
numbers, and the latter are generally greater than necessary for pulling an aircraft out of a spin. Therefore, in pulling supersonic aircraft out of spins, it is necessary in most cases to insure that the aerodynamic moments built up in the escape process do not become too great (danger of "transfer" of the controls during escape). Excess moments during escape can only lead to a considerable deterioration of the escape characteristics (for example, to an extremely steep dive after the autorotation of the aircraft has ceased, along with an increase in loss of altitude during the escape, etc.), and may even lead to failure of the aircraft to escape from spin (transition from normal spin to inverted, from right-hand to left-hand, etc.). Under these conditions, only the "weak" methods (Nos. 1 and 2) can be used for escaping the spin.

However, under certain conditions, supersonic aircraft can fall into regimes of stable spin with very intense rotation of the aircraft, when only the "strong" methods of escape will suffice. This means that in actual practice, when it is desired to pull a modern supersonic aircraft out of a normal spin, both "weak" and "strong" methods should be used. Under no circumstances should it be forgotten that the "stronger" methods do not "overlap" in any way and cannot replace the "weak" methods. Each method of escape has its own range of application.

The selection of the required method of pulling the aircraft out of the spin is determined only by the nature of the particular regime and need not be related to any other parameters (for example, the flight altitude): the latter can be used by the pilot only as auxiliary sources of information and for facilitating and speeding up his correct determination of the nature of the regime. In all cases, the movement of the control surfaces against the spin must be done as forcefully as possible. Slow, sluggish movement of the control surfaces causes deterioration of the escape characteristics and sometimes simply makes it impossible to escape the spin.

The interval between the movement of the rudder and the elevator (movable stabilizer) should be counted by the pilot in seconds rather than in turns, since experience has shown that even a small change in the nature of the regime (for example, a slight slowing down or acceleration of the aircraft's rotation) makes it very difficult and sometimes impossible to count the turns accurately without making serious errors, especially when the aircraft is undergoing significant changes in its position in space (with the aircraft periodically falling on its back, etc.). In addition, the use of the concept "spin turns" is undesirable, not only with respect to escape from spin, but also during the regime, since it tells the pilot very little (in non-uniform spin regime, where the aircraft is falling like a leaf, the idea of "turns" simply loses its meaning). The most important characteristics are the duration of the regime and the loss of altitude by the aircraft during this time.

Counting in seconds is always less difficult and more reliable /210 for the pilots, and it is also a convenient means of determining the interval of time which has elapsed between the movement of the control surfaces in the attempt to escape as well as in estimating the delay in escape. Calculations of these values in seconds is usually done by counting the seconds out loud.

In addition to measuring the duration of the regime in seconds, the pilot must also watch the altimeter during spins. The absolute altitude value as indicated by an altimeter aboard the aircraft shows considerable errors in a spin; these are caused by the partial sensing of the velocity head by the static openings of the air-pressure intake at high angles of attack, as well as by the large angles of sideslip which may appear in spin and the high angular velocities of rotation of the aircraft. However, the loss of altitude in a spin can be estimated quite accurately with the aid of a visual device; however, altitude is one of the most important flight characteristics for the pilot, since it is directly related with the conditions for insuring safety in flight (especially when the aircraft is in spin at a relatively low altitude).

Flying experience has shown that the first two methods of escape are the ones most frequently used by supersonic aircraft. If an initial attempt to pull the aircraft out of a spin fails and autorotation does not stop, the pilot must set the control surfaces against the spin once again and make another attempt at escape in 2-4 seconds. This time however, he should use a "stronger" method since in the first case he presumably did not employ a "strong" method (i.e., he incorrectly judged the nature of the spin).

(c) ESCAPE FROM INVERTED SPIN

The methods of escaping from inverted spin in modern supersonic aircraft are the following (Fig. 5.21):

(1) Method 1 for inverted spin consists of escaping from spin by simultaneously setting the elevator and rudder in a neutral position, with the ailerons in a neutral position.

(2) Method 2 for inverted spin consists of escaping from spin by setting the rudder all the way over against spin and then (after

2-4 sec) moving the elevator to a neutral position, with the ailerons in a neutral position.

(3) Method 3 for inverted spin consists of escaping from spin by moving the rudder all the way over against the spin and then (after 2-4 seconds) moving the elevator all the way against the spin, also with the ailerons in a neutral position.

To pull supersonic aircraft out of inverted spins, Method No. 2 is the one most often used; this is because when such aircraft enter an inverted spin, the latter is usually a stable, oscillating, inverted spin. In general, aircraft fall into inverted spin much more /2 infrequently than the normal one.

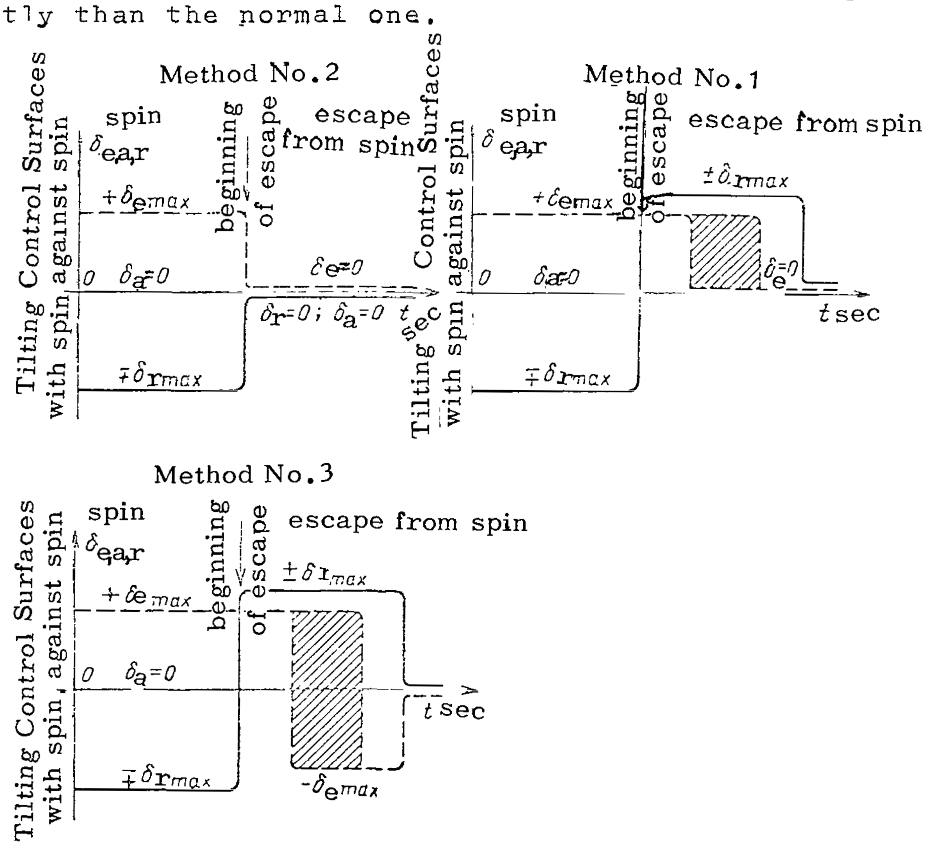


Fig. 5.21. Schematic diagram of the effect of moving the control surfaces in different methods of pulling an aircraft out of inverted spin.

Aircraft of conventional design emerge from inverted spin more easily than they do from normal spin. This is explained by the lower intensity of the aircraft's autorotation at negative subcritical angles of attack, the greater effectiveness of the control surfaces in inverted spin (thanks to the reduced shadowing of the secondary flow over the wing, horizontal empennage and fuselage, as well as the reduction in their effective sweepback angle) and the lower absolute angles for the average angles of attack of such aircraft in inverted spin.

However, regardless of these facts, inverted spin poses considerably more difficulties for the pilot in comparison to normal spin. This is due to the unusual position of the pilot in space during such a regime; he finds himself hanging downward in his harness with a negative force acting on him ($n_y < 0$) which tends to lift him out of his seat. Under such conditions, the pilot may take his hands off the control stick and remove his feet from the pedals, especially if he is not strapped firmly to his seat. Orientation of the pilot /212 in inverted spin is still more complex. When the aircraft is making considerable oscillations, the pilot sometimes finds it difficult to determine visually which type of spin he is in (normal or inverted), especially if the longitudinal axis of the aircraft is close to vertical when the aircraft spins at negative supercritical angles of attack which are low in absolute value. Under these conditions, escape can be achieved by placing the control surfaces in a neutral position, since a situation of this kind usually occurs in regimes of unstable inverted spin.

In some components of supersonic aircraft, the shadowing of the empennage in inverted spin may be greater than in normal spin. This makes it much more difficult to pull such aircraft out of inverted spin and makes it necessary to employ a "stronger" method of escape (Method 3 for inverted spin).

Examples of pulling supersonic aircraft out of inverted spin by the three methods recommended above are shown in Fig. 4.25 (Method 1 for inverted spin) Fig. 4.27 (Method 2) and Fig. 4.30 (Method 3).

If the control surfaces are tilted against the spin when attempting to escape, the pilot must slowly move them to a neutral position as the rotation of the aircraft slows down. Only after the aircraft has obtained a sufficient velocity can the pilot begin (very slowly) to pull the control stick backward in order to pull the aircraft out of the dive.

The methods listed above are sufficient for pulling modern aircraft out of all spin regimes that may occur. When aircraft are developed which differ basically in their design and equipment (supersonic aircraft of the "canard" type or "tailless" aircraft with disturbing moment, etc.) as well as aircraft of conventional design (wings, fuselage, empennage) but with inertial and aerodynamic characteristics which are very different from those in contemporary supersonic aircraft, it will be necessary to use new methods or devices for pulling such aircraft out of spins. This may require some development of the existing methods or even in some cases to the application of new means of pulling aircraft out of spins, such as using a special disturbing moment, different movements of the right and left halves of the stabilizer, shifting the exhaust from the engine laterally to stop the aircraft from rotating, using special anti-spin devices (parachutes, rockets), etc.

5.3 Basic Features of Pulling Supersonic Aircraft Out of Spins.

As we have already mentioned, the diversity of forms and instability of characteristics in spin involving supersonic aircraft (usually, a given aircraft can exhibit different spin regimes depending on the initial conditions and characteristics of its pilotage in the regime) lead to the fact that in order to pull the aircraft reliably out of the spin it is necessary to use some standard or universal method of escape.

As mentioned earlier, the great length of the transitional section of the spin (especially at high altitudes) in supersonic aircraft means that regimes of vertical spin are encountered relatively rarely. Due to the long duration of the transitional section of the spin, especially following stalling at high altitudes, pilots of supersonic aircraft must deal primarily with this transitional section when they fall into a spin and not with a regime of stable vertical spin. Usually the pilot begins to pull the supersonic aircraft out of the spin (and does so) by operating his controls properly, long before the vertical spin begins. /210

If for some reason the spin regime is prolonged and the supersonic aircraft goes on to enter a vertical spin, escape from the latter under certain conditions may be very difficult. To pull a supersonic aircraft out of such a regime, Method 4 can be employed. This is due primarily to the large dispersion of masses in the direction of the longitudinal axis in supersonic aircraft (large inertial moments of pitch).

As we have already pointed out, the four characteristics of escape (hampered escape) from spin were encountered much more frequently in subsonic aircraft, especially in old subsonic aircraft like the I-153, UT-2, etc. However, in those aircraft the poor escape from spin was caused mainly by another factor, the low effectiveness of control surfaces in this regime. In addition, due to the brief duration of the transitional section, the pilots of these subsonic aircraft were required to pull the aircraft out of a vertical spin (the transitional section was over before the pilot could orient himself in the regime).

Pulling the aircraft out of a spin in the transitional section does not usually require "strong" methods; as a rule, Methods 1 and 2 will suffice for normal spin. In this case, the aircraft escapes easily from the spin, due to the low stability or even instability of the regime. Therefore, when a pilot finds his aircraft falling into a spin, he must immediately set the control surfaces with the spin and move the ailerons to a neutral position; as soon as he has succeded in determing the nature of the regime, he must begin the escape process, not allowing the aircraft to enter a stable regime of vertical spin if possible. The transition of the aircraft into a regime of vertical spin at the end of the transitional section usually takes place within a short period of time. This absolute necessity to make a rapid and very significant change in the nature

of the regime (the occurrence of a more stable spin) is unpleasant for the pilot, especially if he has just begun the process of escaping from the spin, having decided to use a relatively "weak" /214
method of escape. Under such conditions, he must be able to estimate the nature of the new regime correctly and to escape by using a "stronger" method.

When the aircraft enters critical or nearly critical flight regimes, the pilot must act as follows: when the aircraft assumes near critical angles of attack and the first signs of stalling appear, the pilot must take all measures necessary to prevent stall. If he does not succeed, and stall occurs, he must try to pull the aircraft out of the stall without setting the control surfaces immediately with the spin; this would cause the aircraft to go into a spin spontaneously, and it would be more difficult to escape from the spin than it would be to escape from a stall. If this also fails and the aircraft enters a spin, the pilot must immediately move the control surfaces into the spin and then (as soon as he has oriented himself in the regime and selected the proper moment for escape) he must immediately begin to escape from the spin. In some cases, in order to facilitate his orientation in the regime, an inexperienced pilot may wait until a more uniform regime of spin is established (orientation of the pilot in the unstable transitional portion of spin, especially at high altitudes, is often more difficult than under the more uniform vertical regime). This is completely wrong, since it may result in the aircraft going into a highly stable regime of spin from which it is still more difficult to escape.

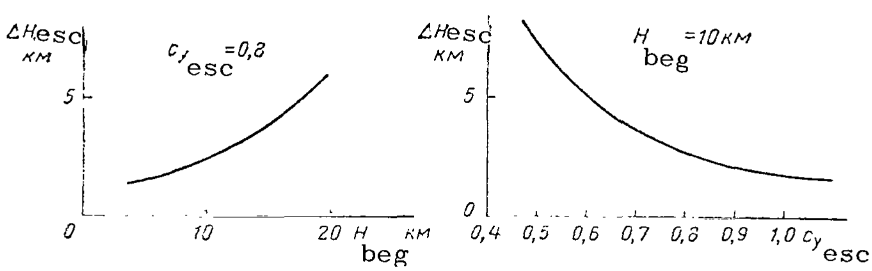


Fig. 5.22. Loss of Altitude During Escape from Normal Spin as a Function of the Coefficient of Lift and the Altitude at Which the Spin Begins.

The decrease in lift and especially the reduction in $c_{y_{per}}$ leads to a greater loss of altitude when a supersonic aircraft emerges from a spin. Fig. 5.22 shows an example of the change in the loss of altitude during escape ΔH_{esc} as a function of the value of coefficient of lift during escape $c_{y_{esc}}$, as well as a function of the altitude at which escape begins H_{beg}, for a supersonic aircraft. To reduce the loss of altitude during escape, the aircraft must be piloted so that the maximum use is made of its lift, while making sure at the same time that the aircraft will not enter a second stall or exceed the permissible limits for stability of normal force, i.e., the values $c_{y_{per}}(\alpha_{per})$ and $n^{a}_{y\,max}$ must not be exceeded under any circumstances. With this goal in mind, it is desirable to have aboard /2 the aircraft suitable visual indicators and signaling systems which would include in particular the device discussed earlier (in Chapter III), the two-vane indicator of angle of attack and the lateral-load factor indicator.

The methods recommended for pulling an aircraft out of a spin do not require the pilot to make a definite choice of a certain moment for beginning his escape procedure (the moment when he must move the first control surface to escape). However, test pilots always select the moment (for example, when the nose falls or when rotation stops) which ensures optimum escape of the aircraft from the spin, as rapidly and safely as possible, with minimum delay and loss of altitude. In order for the pilot to determine the features of each concrete type of spin rapidly and correctly and to select the proper method of escaping from this regime as well as to select the optimum moment to begin his escape procedure, he must study the characteristics of spin for his particular aircraft, in other words, he must be trained for a spin.

In preparing a program for training pilots, it is necessary to remember that these recommendations can only be carried out under conditions where the pilot has been well trained in going into a spin and escaping from it. Training pilots for spins enables them to fly more skillfully at critical regimes and to "squeeze" out of the most it can give. To ensure complete safety when training for spins, the first stages of such training should include putting the aircraft into a spin (for training purposes) from an initial regime of straight-line horizontal flight at a speed greater than a minimum of 30-60 km/hr. Stall must be produced by a vigorous movement of the rudder, with a simultaneous sharp pull of the control stick backward. This method produces a definite stall and will cause the aircraft to go into a spin in a certain direction of rotation. The training of pilots for spin must be carried out in two-seated aircraft.

5.4 Piloting Errors in Recovery From Spin

In pulling an aircraft out of a spin,pilots who have not received sufficient training in such regimes are most likely to make two errors: they may tilt the ailerons against the spin and employ a "stronger" method of escape than necessary in view of the characteristics of the given regime. Tilting the ailerons against the spin in supersonic aircraft, as mentioned above, usually leads to a considerable complication of the escape process and sometimes makes it impossible.

Unnecessary use of a "strong" method of escape (overcautiousness) not only does not increase the reliability of escape technique, but on the other hand merely makes the characteristics of escape more difficult; sometimes it may even make it impossible for the pilot to escape from the spin. Under these conditions the failure to escape does not mean that the aircraft continues to rotate in a certain spin regime, but rather that the aircraft makes a transition from this regime to another regime, for example from right-hand to left-hand, or from normal to inverted spin, etc. In other words, the first spin regime ends, but a second begins which is opposite in direction to the first or has reversed sign for the average angles of attack of the aircraft, i.e., the aircraft continues to spin but in another regime. /216

(a) CONTRIBUTION OF EXCESS MOMENTS TO ESCAPE

Tilting the rudder during escape all the way over against the direction of spin, when the characteristics of the regime have revealed that it is sufficient merely to hold it in a neutral position, may cause a considerable increase in the time required to escape or may make it impossible; the aircraft simply changes the direction of rotation (the sign of ωy) but does not emerge from the spin. Thus, Fig. 5.23 shows an example which illustrates an attempt on the part of the pilot to attempt to pull his aircraft out of an unstable right-hand normal spin, which occurred in the form of a downward drift along a spiral trajectory with the rudder tilted all the way over (at the moment $t \approx 43.5$ sec, the rudder was set to $\delta_r \approx -20°$), so that the nature of the regime indicated that escape from it would require moving the rudder only to a neutral position (using Method 1 for escape). As a result, the aircraft did not escape from the spin, but simply shifted from right-hand to left-hand normal spin (beginning at $t \approx 45$ sec), from which it was then withdrawn by having the pilot set the rudder in a neutral position ($t \approx 50$ sec).

The movement of the control stick all the way forward in escaping from normal spin, when the characteristics of the regime indicate that it is adequate simply to hold it in a neutral position, can cause the aircraft to shift to a regime of inverted spin (see Fig. 4.30), i.e., it will not escape from the spin or else the nose of the aircraft will fall excessively when autorotation stops. The latter produces an increase in the initial angle of dive and therefore a steeper dive in which the loss of altitude and final diving velocity increase; this is particularly dangerous at low altitudes, due to the fact that such a dive may exceed the permissable instrument flight speed.

In addition, the increase in the absolute angle of dive produces an increase in the loss of altitude for escape, which in supersonic aircraft (even when correctly piloted) is quite high (see Fig. 5.22). The production of overly high diving moments when escaping, due to the control stick being pushed too far forward, also causes large negative load factors (in Fig. 4.31 at $t \approx 55$ sec the negative normal load factor reached a value of $ny \approx -1.7$).

Physiological tests which have been carried out indicate that the influence of negative load factor has a much greater effect on the activity of a pilot than does a positive normal load factor on the same /217 absolute value. This means that the coordination of the pilot deteriorates, and it is more difficult for him to judge his position in space, etc. The effect of a negative normal force may have other very unpleasant effects: pulling the pilot out of his seat, breakage of the safety harness, striking the head against the canopy, etc.

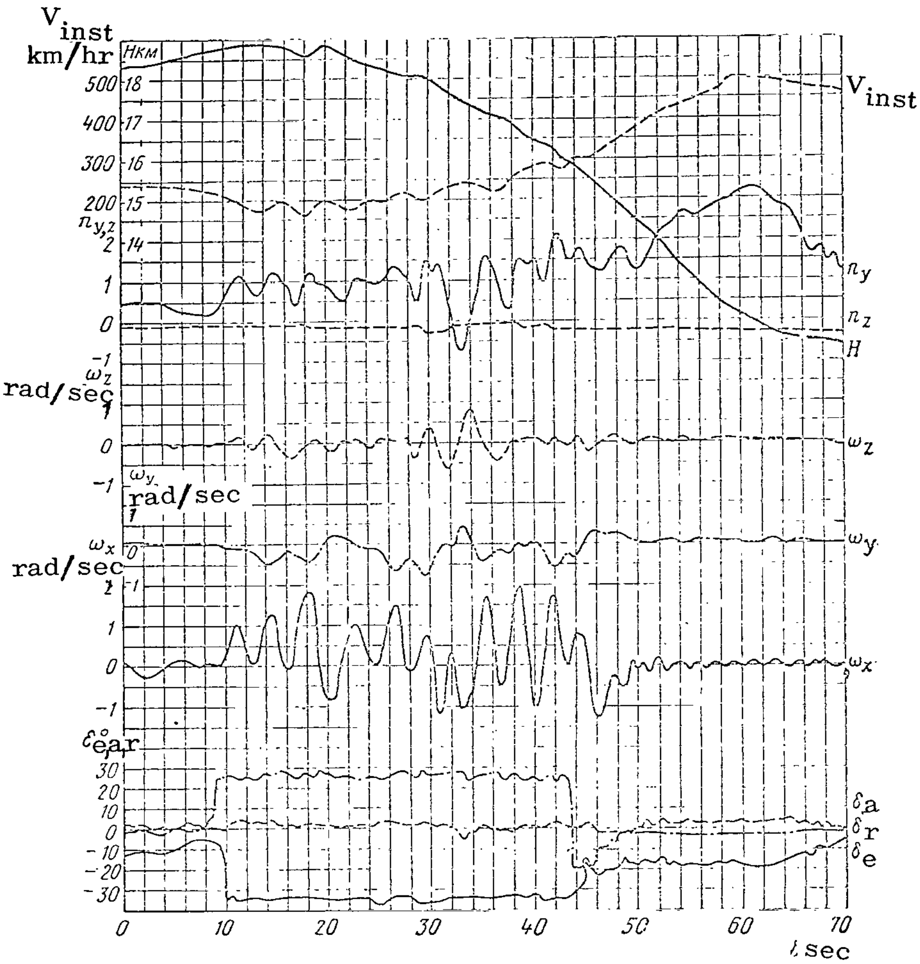


Fig. 5.23. Example of Piloting Error: Excessive Movement of the Rudder ($t \approx$ 43.5 sec; $\delta_r \approx -20°$) in Escaping From Spin.

The use of an overly "strong" method of escaping from spin requires the pilot to make more rapid and skilled motions with the control surfaces, since more excessive aerodynamic moments are involved. In this case, the aircraft becomes more "rigid" with respect to piloting errors, (for example, with respect to the use of the ailerons).

The examples which we have given illustrate the important properties listed above for pulling supersonic aircraft out of spins;

the difficulties in pulling supersonic aircraft out of a spin in the majority of cases creates problems which have nothing to do with the fact that the control surfaces do not suffice for escape, as was sometimes the case in old aircraft, but rather the fact that an inexperienced pilot does not know how to select the proper method of excaping from a given regime.

Such piloting errors have also been observed in subsonic aircraft as well, but the danger and possibility of their occurrence is considerably greater in supersonic aircraft. The latter is related to the large number of different spin regimes, as well as the expansion of the range of methods employed for escaping from a spin in supersonic aircraft. Thus, for example, a given aircraft under different conditions may exhibit regimes of normal spin which require the use of all four methods of escape (see Fig. 5.20): each regime has its own method of escape.

(b) MOVEMENT OF CONTROL SURFACES IN THE REVERSE SEQUENCE

A type of error in pilotage which must be avoided is tilting the elevator for escape before moving the rudder (so-called reverse sequence of tilting the control surfaces when pulling an aircraft out of a spin). An error of this kind usually makes it impossible to pull the aircraft out of a spin. Under no circumstances must the elevator be moved before the rudder when pulling out of a spin. The elevator must be operated to escape the spin only after a certain interval of time has passed after moving the rudder, or else it must be done simultaneously with moving the rudder (in unstable spin).

This is due mainly to the reduction of inertial moment of pitch $M_{z\,\mathrm{in}} = (J_x - J_y)\omega_x\omega_y$ after the rudder has been moved for escape; this is caused by a drop in the absolute value of the moments, and therefore of the angular velocity of autorotation of the aircraft as a result of the appearance of an inward sideslip. The latter is produced by moving the rudder against the spin when pulling the aircraft out of a stable regime.

As mentioned earlier (in Chapter IV), this inertial moment of pitch involves oscillation. Therefore, its reduction leads to a drop in the aerodynamic moment of dive required for the aircraft to escape the spin (a shift to a subcritical angle of attack), produced by movement of the elevator. Hence, moving the control surfaces in the proper order in order to escape the spin (tilting the elevator after moving the rudder to escape) reduces the necessary work of the elevator and simultaneously makes it easier to pull the aircraft out of the spin, since it also makes it easier for the elevator to operate when shifting the aircraft into subcritical angles of attack.

Reduction of the inertial oscillating moment also causes the nose of the aircraft to drop, thus reducing its angle of attack in the spin. This means that the effectiveness of the elevator can increase considerably. This produces greater aerodynamic moments of dive for the aircraft which is attempting to escape from spin, i.e., /219 there is still one more factor which facilitates the aircraft's escape from spin when the proper sequence for moving the control surfaces is followed.

Of equal importance is the fact that the moment of dive which is produced by moving the elevator during the process of escape reduces the angle of attack, while at small angles of attack the static stability of the aircraft along its path is usually increased (or developed for the first time). As a result, it becomes more difficult to use the rudder to create the inward sideslip which is desirable from the standpoint of insuring escape from spin (the effect of the elevator, as a rule, is insufficient to shift the aircraft to subcritical angles of attack).

Moving the elevator (and especially the movable stabilizer) to escape spin before moving the rudder can also lead to a considerable increase in the shadowing of flow over the horizontal part of the empennage, thus reducing the effectiveness of the rudder.

Waiting the required period of time between moving the control surfaces and pulling an aircraft out of a stable spin is necessary in order that the rudder which has been tilted can generate the required inward sideslip, which in turn must be sufficiently effective in reducing the absolute value of the autorotational moment. When pulling an aircraft out of an unstable spin, the control surfaces are tilted simultaneously to escape, since in this case even a small increase is sufficient to terminate the regime, while sometimes a spin of this kind can only exist with the control surfaces tilted with the spin.

As we have already pointed out, an increase in the degree of static stability along its path, which usually arises when the angle of attack is reduced (except in those cases when spin is occuring at very high or supercritical angles of attack, when the aircraft may have a considerable degree of stability along its path; see for example Fig. 4.46), it becomes difficult to create inward sideslip in the case where the rudder is moved after the elevator has been moved. This can be one of the most important reasons for causing deterioration of conditions for escaping from spin when using the reverse order of moving the control surfaces to escape. However, we must keep in mind the existence of the following contradictory factors when discussing the reason for this phenomenon. On the one hand, the instability of the aircraft along its path, which usually appears at large angles of attack, causes an increase in the absolute values of the sideslip angle and in the angular velocity of yaw of the aircraft in spin. The latter causes an increase in the inertial moment /2
$M_{z\,in}$, which in turn causes an increase in the angle of attack of the aircraft and an increase in the aerodynamic diving moment which is required for the aircraft to escape from the spin. The greater the positive derivative m_y^{β}, the greater the sideslip angle will be and the larger the angular velocity of yaw and the angle of attack must be.

When the rudder is tilted with spin and the aircraft is directionally unstable, angles of outward sideslip may be formed which are larger in absolute value than is the case for stable regimes. As a result, the changes in the sideslip angle which are required to produce inward sideslip (to pull the aircraft out of the spin) increase. All of this makes it more difficult to pull the aircraft out of the spin.

On the other hand, however, the directional instability of the aircraft causes an increase in the variations of the angular velocity of yaw and the sideslip angle of the aircraft. These oscillations increase sharply with an increase in the degree of directional instability and a reinforcement of the nonlinear character of the curve of its dependence on the angles of attack and sideslip (which usually occurs at supercritical angles of attack).

The changes in the angle of attack and the angular velocity of yaw mean that the influence of the transverse static stability of the aircraft (the effect of the restoring moment of roll $\Delta M_{x\beta} = M_x^\beta \beta$ and the spiral moment of roll $\Delta M_{x\omega_y} = M_x^{\omega_y} \omega_y$ produces intense oscillations in the angular velocity of roll. The changes in the angular velocities of roll and yaw under the influence of the inertial moment of pitchs. $M_{z\,\text{in}}$ caused changes in the velocity of pitch and consequently in the angle of attack of the aircraft. Therefore, an increase in the directional instability produces a sharp increase in both the forward and transverse (as well as longitudinal) oscillations of the aircraft.

With considerable directional instability of the aircraft, the changes in the angle of attack during spin may increase to the point where the aircraft will periodically approach subcritical regimes and in some cases will periodically slip into these regimes, with the control surfaces held firmly in place with the spin. However, due to the great variations in the angle of sideslip, there may be a periodic occurrence of a considerable inward sideslip which significantly reduces the absolute value of the autorotational moment of the wings. All of these factors make it easier to pull the aircraft out of the spin.

(c) TILTING THE AILERONS

Another inadmissable error in piloting is movimg the ailerons in the regime, especially when escaping from a spin (except in the case when Method 4 is used). The pilot must attempt to keep the ailerons in a strictly neutral position during the spin and when escaping from it (or in an initially balanced condition if it is different from the neutral position). Even a small tilting of the /221
aileron is usually sufficient to cause significant changes in the nature of spin, and therefore in the conditions for escaping from it. Fig. 5.24 shows a regime in which the ailerons were tilted against spin to an angle of $\Delta\delta_a \approx 2\text{-}3°$ (an error which a pilot can easily make if he does not pay sufficient attention to the position of the control stick during the regime). This has been found to be

sufficient so that following an unintentional, very brief movement of the control stick, the aircraft changed from a left-hand normal spin to a left-hand inverted spin ($t \approx 25$ sec). The pilot was unable to orient himself to this change in the regime and continued to hold the control surfaces in a position corresponding to left-hand normal spin. This meant that the inverted spin was continued and he escaped from it only by setting the control surfaces in a neutral condition.

A very serious error in piloting which usually makes it impossible to pull the aircraft out of the spin is moving the ailerons against the spin when attempting to escape (in supersonic aircraft, this usually only speeds up the autorotation).

(d) DIFFICULTY IN DETERMINING THE NATURE OF SPIN AND THE POSITION OF THE AIRCRAFT IN THE SKY.

In a number of cases, as we have already pointed out, errors in piloting can be committed due to difficulty in the pilot's orienting himself in spin or (more exactly) due to an incorrect estimate on his part of the nature of the motion, the direction of rotation, and the position of the aircraft in space. In the example shown in Fig. 5.25, the pilot found it difficult to orient himself in the regime (i.e., to estimate the nature of the regime, so that he did not realize the aircraft had shifted from a left-hand normal spin to a left-hand inverted spin at $t \approx 19$ sec): as a result, he continued trying to pull the aircraft out of the inverted spin by using Method 2 for escape from normal spin. This error caused the aircraft to shift from left-hand inverted spin to right-hand normal spin, for which the corresponding movement of the rudder sufficed to place it in a spin. Escape from spin took place only after the controls were set to a neutral position.

Setting the controls in a neutral position to escape is the best thing the pilot can do under these circumstances, when he is unable to orient himself as to the nature of the spin. This is because such a difficulty in orientation in a properly trained pilot can only occur under highly unstable conditions of spin, involving considerable oscillations of the aircraft, i.e., in an unstable regime. However, in this case, it is also necessary to employ Escape Method 1.

(e) USE OF AN INSUFFICIENTLY "STRONG" (EFFECTIVE) METHOD OF RECOVERY

We mentioned earlier that the choice of a method of pulling an aircraft out of a spin is made in accordance with the nature of the regime. This means that it is necessary to avoid methods of escape which are either "too strong" (overcautiousness) or insufficiently "strong" (when the pilot is unable to estimate the intensity and stability of the aircraft's rotation in the spin). Using a "weaker" method of escape than is required for a given spin naturally means that the aircraft fails to emerge from the spin, or that it does

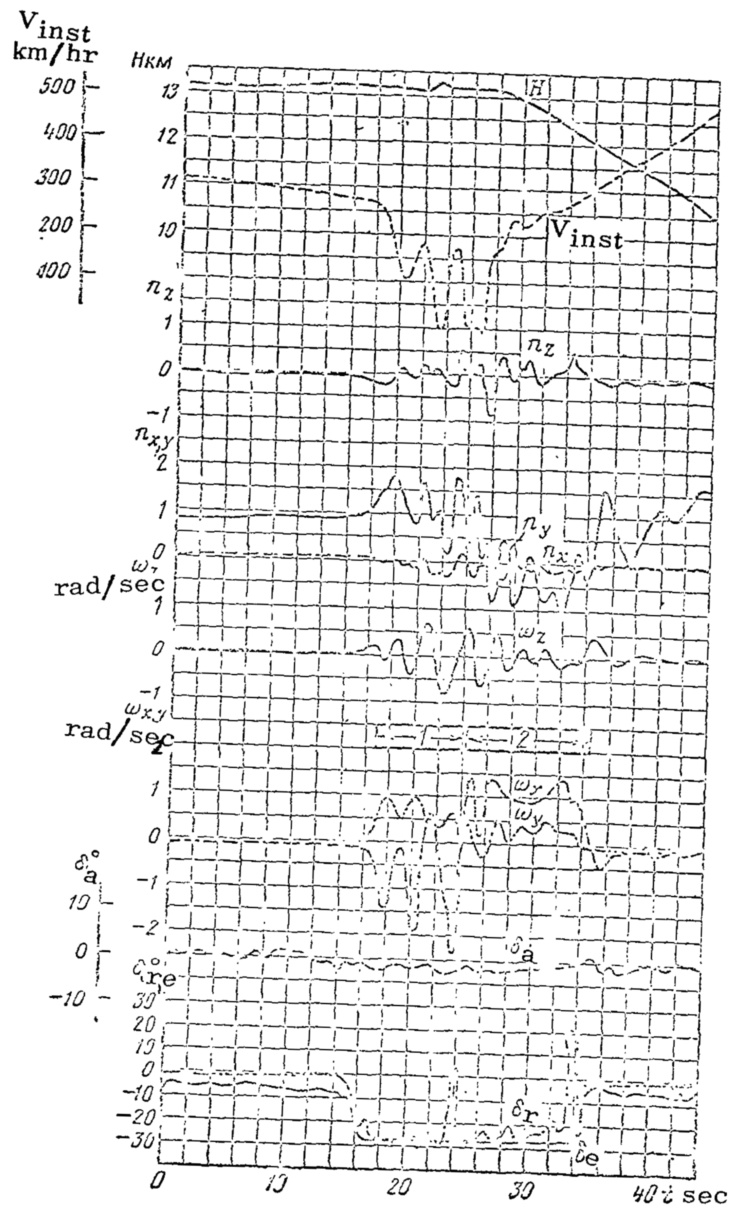


Fig. 5.24. Example of Piloting Error Sending an Aircraft Out of Normal into Inverted Spin: (1) Left-Hand Normal Spin; (2) Left-Hand Inverted Spin.

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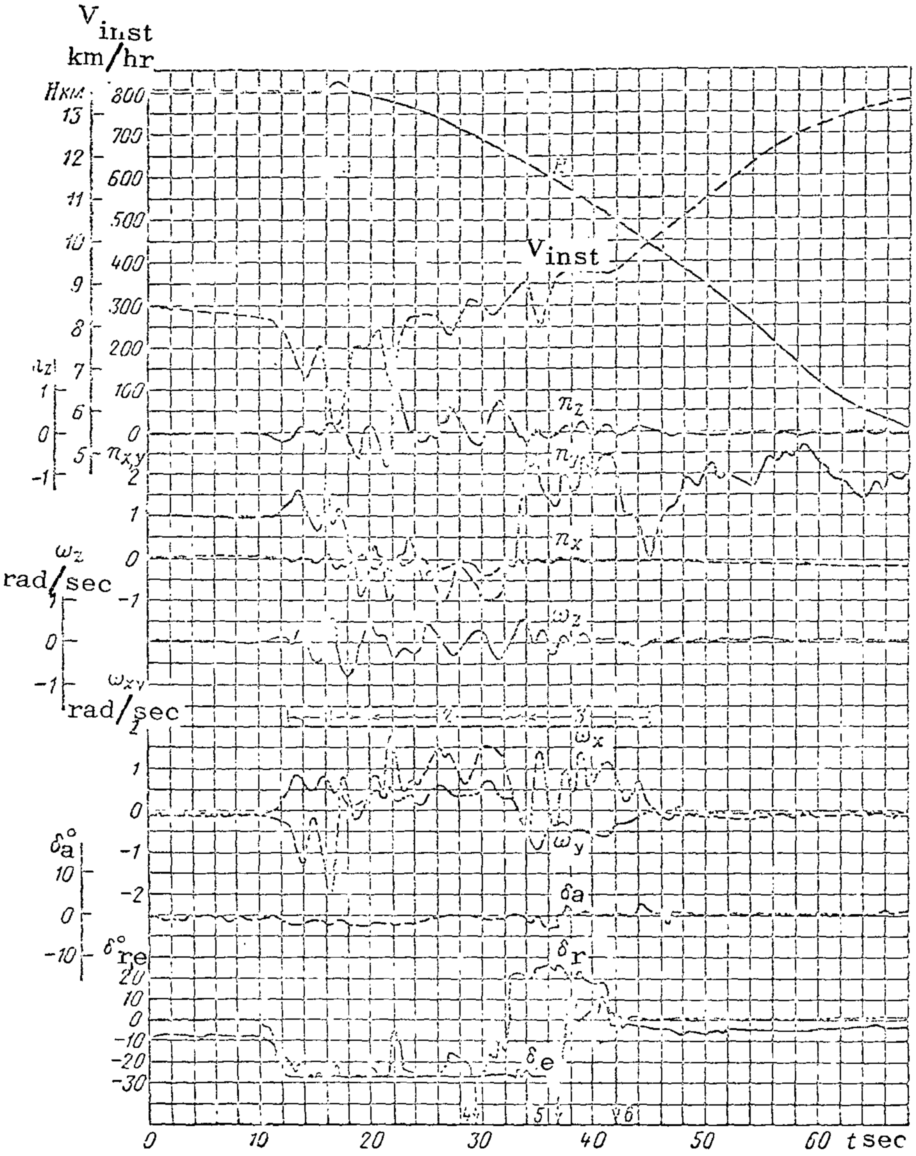

Fig. 5.25. Example of Piloting Error Produced by Difficulty in Orientation on the Part of the Pilot (His Estimation of the Nature of the Aircraft's Motion) During Spin: (1) Left-Hand Normal Spin; (2) Left-Hand Inverted Spin; (3) Right-Hand Normal Spin; (4) Beginning of Movement of Rudder to Escape From Spin; (5) Beginning of Movement of Elevator for Escape from Spin; (6) Moment When Both Control Surfaces Were Practically in a Neutral Position.

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escape but with a considerable delay. The latter circumstance can actually be viewed as failure to escape, since from a practical standpoint, a very prolonged period of rotation of the aircraft following movement of the control surfaces to escape is taken by the pilot as failure to escape. /224

Use of an insufficiently "strong" method of escape is inadmissible, not only in the case of stable spin with uniform and intense rotation of the aircraft, but also in the case of unstable spin, due to the increased delay in escaping. At low altitudes this can be dangerous for two reasons: in the first place, due to the considerable increase in the delay itself, and secondly, due to the loss of altitude available for escape when an aircraft slips into a spin at low altitude.

In pulling the aircraft out of a spin, the pilot must always wait as long as possible after moving a control surface to escape (at least 12-15 seconds, and longer when sufficient altitude is available) before moving these surfaces again with spin, beginning a second attempt to escape by using a "stronger" method if the first attempt to escape has not been successful. This is because, in the first place, the escape of the aircraft can take place with a relatively long delay, (particularly as the result of piloting errors which are usually not so serious that they make it impossible to pull the aircraft out of the spin), and in the second place because of the excited state of the pilot in this unusual (non-operational) flight regime for periods of time lasting several seconds, during which time he may think a much longer period of time has passed.

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